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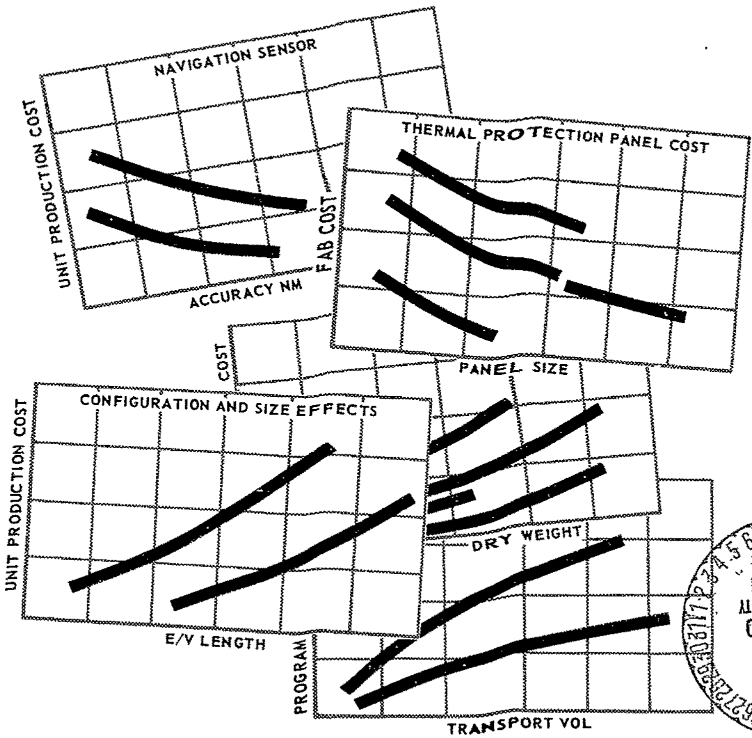
OPTIMIZED COST/PERFORMANCE DESIGN METHODOLOGY



VOLUME III CONCEPT ANALYSIS AND MODEL DEVELOPMENT

BOOK 2 - MODEL FORMULATION

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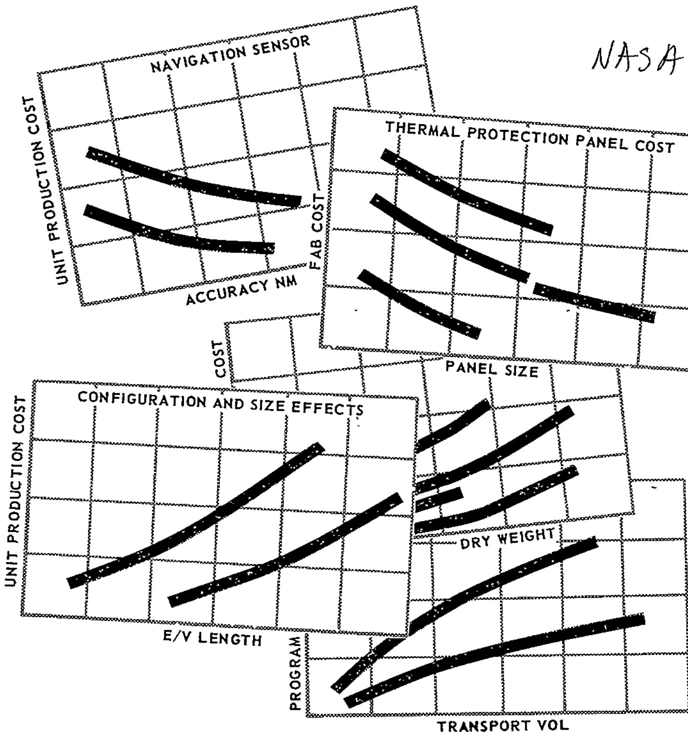

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FOREWORD

This report is submitted to NASA, the Mission Analysis Division of OART, as part of the final reporting on Contract NAS 2-5022, Optimized Cost/Performance Design Methodology of Orbital Transportation Systems. This twelve month study was initiated in July 1968 and was performed in two general phases: a data review and analysis phase and a system evaluation phase. The reporting of the study is organized in three volumes but includes several books in Volumes 2 and 3. Volume 1 is a short summary of the complete study, Volume 2 covers the phase 1 data review and analysis, and Volume 3 covers the phase 2 system evaluation. The Study Manager was L. M. McKay; the major Task Leaders were P. T. Gentle, V. E. Henderson, L. E. Smith, and A. D. Trautman. The NASA Technical Monitor was C. D. Havill.

McDonnell Douglas gratefully acknowledges the support and cooperation of many companies which supplied information to the study. A list of the companies and their area of contribution is included in Volume II, Book 1, Appendix A.

ABSTRACT

The broad objectives of this study were to gather historical cost and performance data, organize and analyze the data so that cost estimating relationships could be developed, and evaluate several system concepts for space logistics support.

The primary source of historical cost data was the Gemini and Saturn Programs and cost estimating relationships draw extensively on this experience. A range of reuse concepts were evaluated and optimum (least cost) concepts defined for a variety of program options. These include variations in such things as crew size, cargo capacity, program requirements, etc. for either ballistic or lifting body (M2-F2) entry vehicles.

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1.0 SUMMARY - This optimization program relates various operating and design parameters to cost and, by using search and optimization techniques, finds the least cost system for the specified parameters. The logic that was developed to do this is presented in this book.

There are three major modules to this model. The first is the mair module which is the executive control logic. The second is the size module which translates performance and operational requirements into a vehicle description and weight statement. The third module is the cost module which develops the total program costs from the data supplied by the other two modules. The executive and the size modules are each composed of several subroutines which are called for as required.

The main module contains the executive control subroutine, inventory subroutine, flow time subroutine, the comparison logic and the optimization logic. The sizing module contains its own control logic and crew size, geometry, aerodynamics, thermodynamics, power, ECS, mass properties, loads, structure, propulsion, landing, size and output subroutines. The cost module develops spacecraft first unit costs, development costs, investment costs, and operational costs and launch vehicle development, investment and operational costs.

At present, this model is limited to two types of spacecraft - ballistic and low L/D lifting body shapes. The model can be easily expanded to other types such as medium or high L/D lifting body shapes with or without variable geometry wings. Provision to handle self-launching vehicles can be added without extensive modification. The launch vehicle portion is very brief and parametric, and provides reasonable order-of-magnitude costs only. A better launch vehicle sizing and costing model is needed to make this model truly a cost/performance optimization model.

1.1 TOP LEVEL LOGIC FLOW - The logic flow for this program is shown in Figure 1.1-1. The user specifies the required inputs to initialize the program. The main module controls the call up of the necessary modules based upon the instructions the user has input. The first module called is the sizing module which develops a vehicle description and weight statement. Next, the flow time subrouting is called to determine the time required to cycle reusable entry vehicles. This time is utilized by the inventory subroutine in determining the quantity of vehicles which must be procured to fulfill the mission requirements specified. The inventory subroutine considers probability of mission success, expected losses, and turn around time in arriving at the total number of units necessary.

The cost module is called next where the first unit spacecraft costs are estimated based upon the description developed by the size module and the user inputs. Then it determines the development, investment and operational costs for both the spacecraft and the launch vehicle.

If the user has specified that an optimum size and/or subsystem mix be determined, the main module specifies a new set of initial data and calls for the size module again. The results of each pass through the various modules and subroutines are stored by the main module until the desired minimum has been found. In essence, the program tries all possible alternatives and then selects the one having the lowest total program cost. This is the lowest cradle-to-grave costs which considers both development and operational costs.

1.2 SECOND LEVEL LOGIC FLOW - The size module and the cost module are both rather complex. Each can become a stand-alone module which will operate and produce satisfactory results independent of this program or the other modules.

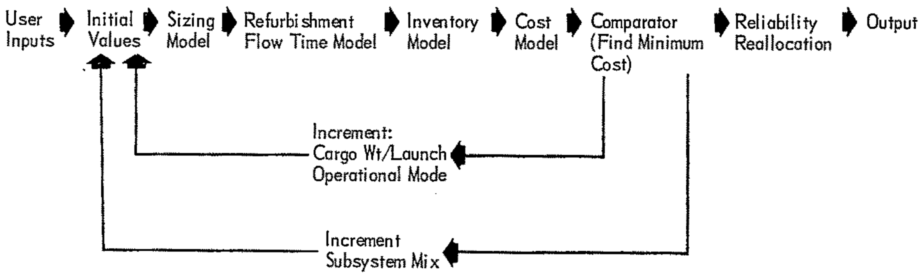
1.2.1 THE SIZE MODULE - The logic flow of this module is shown in Figure 1.2-1. This shows how this module develops the description and weight statement from parametric data and iteration. These are several iterative loops within this module before it converges to an answer which does not violate any of the constraints which are built in.

1.2.2 THE COST MODULE - The logic flow of this module is shown in Figure 1.2-2. After the S/C first unit costs are calculated, the design and development costs are computed. The ground test, flight test and air drop hardware costs are determined, and the test operations costs estimated. The support and AGE costs for the development phase are calculated.

Next, the production costs are computed and then the operational costs such as launch operations, launch area support, recovery, refurbishment, etc., are calculated. Management costs and fee is added to provide a total program cost.

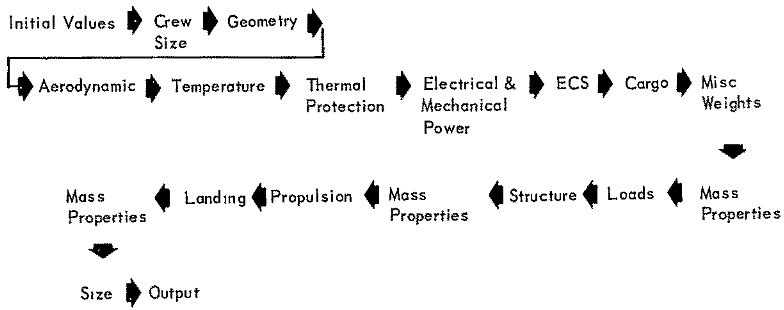
FIGURE 1.1-1

PROGRAM LOGIC



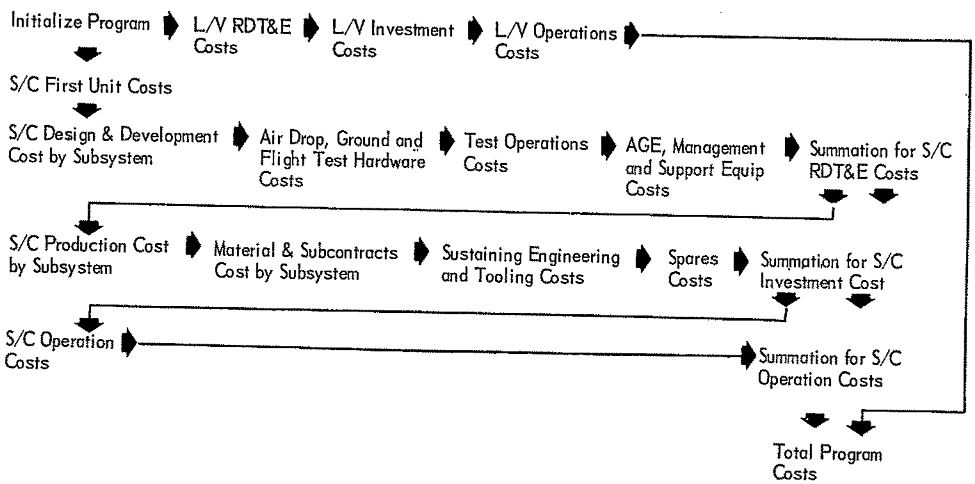
SIZING MODEL

FIGURE 1.2-1



COST MODEL

FIGURE 1.2-2



2.0 INTRODUCTION - The purpose of the Optimized Cost/Performance Design Methodology study was to provide a method of using cost as a basic design parameter in identifying and defining more economical space transportation systems. This study was performed in six tasks as shown in Figure 2-1. Task 1 involved developing the cost data, organizing the data by categories, and developing cost estimating relationships. Tasks 2 and 3 developed the requirements and the physical and functional characteristics of the alternate spacecraft subsystems and operations. An analytical cost model was formulated in Task 4. Task 5 developed the logic, data, and methods for systematically varying the design and operational specifications of each vehicle configuration. Task 6 took all the data and tools developed in the other tasks and then determined the economically optimum design and operational philosophies, sensitivities to program size, launch rate, payload size, the problem areas and technology limitations, for the vehicles being considered.

This book reports on the work accomplished in Tasks 4 and 5. The objectives of these tasks were to:

(1) Adapt an existing sizing model to the requirements of this study and program.
(2) Develop the necessary optimization scope and logic to accomplish the purpose of the study.
(3) Develop the subroutines and models necessary to implement the logic developed.
(4) Perfect the methods for varying the vehicle or operational specifications systematically.
(5) Provide data outputs in suitable form and format to give necessary visibility to the user.

STUDY TASK FLOW

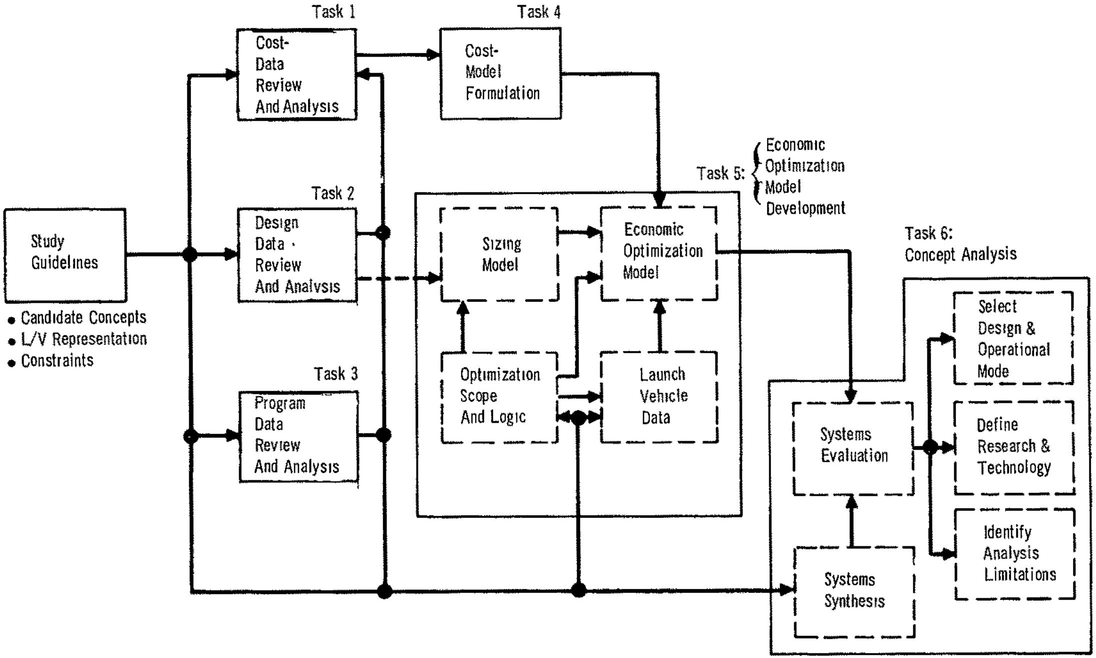


FIGURE 2-1

3.0 SCOPE AND LIMITATIONS - The program provides the user with a versatile tool which has been developed to provide the broadest possible capability within the time and budget limitations of this study. The user has many options available to him, both in the level of detail to be considered and in amount of output data generated. The limitations are those imposed by vehicle configurations and actual computer running time. We have attempted to minimize the latter by making maximum use of the user as a man-in-the-loop element of this program.

3.1 OPTIONS AVAILABLE TO THE USER - The user has operational options, configuration options, performance options and output data options he may select for any particular run he wishes to make. In the operational options he can specify an ETR or WTR launch, the type of AGE equipment, the launch operations philosophy, the refurbishment philosophy, the refurbishment site, the recovery mode, the number of recovery sites, and the transportation mode.

Configuration options include type of vehicle (ballistic or low L/D lifting body), degree of modularity, degree of reuseability, cargo weight per launch, fixed total spacecraft gross weight (booster throw weight limitations), type of subsystem variations to consider as alternatives, and type of launch vehicle to be considered. Performance options are related to configuration, but include added maneuver ΔV, type of propellants and engines, and types of materials to be used in the vehicle.

Output options provide a four page summary of the vehicle description and weights with a one page cost summary as one extreme, to a full 54 page printout of the complete description, weight breakdown, and cost breakdown to the subsystem level. There are two levels of detail for the weight and size data and three levels of detail for the cost data. This provides the user with several options depending on the data visibility he desires.

3.2 Limitations - There are three major limitations to this program: (1) only a ballistic or low L/D lifting body spacecraft can be analyzed, (2) the user can vary some of the operational parameters, but changes to the basic programming are very difficult to make, and (3) a full optimization run will require a long computer run time.

The configuration limitations are a result of the initial study guidelines which specified that ballistic and M2/F2 shapes were to be the basis of the model. Other configurations can be added since the basic size module logic is applicable to any configuration. It requires new geometric descriptions and specification of all of the input matrices.

The user has extensive control over some of the operational parameters as mentioned previously, but these are the only options available to him. He can not delete any of the activities or ground test hardware requirements without going into the basic program which is difficult. An attempt has been made to provide the user with a wide capability but the provided options are the only ones allowable without modification to the program. However the printout visibility is sufficient for the analyst to examine virtually any condition he desires. Coefficients of cost CER's can be changed relatively easily, but the cost equations cannot be changed by adding new variables without major changes to the program.

Computer run time for a single pass through this program (i.e. one cargo size and one cost), requires about 0.6 minutes on a IBM 360-75 machine. A golden rule optimization to an optimum cargo size for a minimum total program cost requires up to fourteen passes and about 9.0 minutes on the IBM 360-75. A full subsystem optimization requires more than thirty passes and 34.0 minutes machine time. Other machines may require more time depending upon their capabilities as compared to the IBM 360-75 system.

4. OCPDM INPUT AND OUTPUT DESCRIPTION

4.1 Data Input Procedure - The OCPDM program requires user inputs for the three basic modules:

a) cargo weight and operational variations optimization
b) subsystem optimization
c) subsystem reliability evaluation

In selecting the cargo weight the user has three basic options available:

1. Fixed Cargo Weight/Launch - The entire S/C is sized around the cargo requirement.
2. Fixed Launch Vehicle Size - Cargo weight is sized to fit the fixed spacecraft size.
3. Optimized Cargo Weight/Launch - Golden Rule is employed to obtain the optimum cargo weight/launch.

The user then has the capability of selecting up to 10 sets of operational variations. Next the user may run various alternates on any combination of the Primary Structure (E/V and Mission Module), Propulsion (Upper Stage Boost and Orbital Maneuver System), Power (Electrical and Hydraulic), Avionics (Guidance, Control and Telecommunications), Thermal Protection and Environmental Control Systems.

Finally the user may elect to run the Subsystem Reliability Model.

A detailed description of all the options is given in section 5.3 of this book.

Data is input to the program via the NAMELIST statement. The following general rules apply when using the namelist statement.

1. The first card must have an ampersand & in card column 2 followed by word INPUT followed by a blank. Then, starting in card column 9, data is coded in the form VARIABLE = value, VARIABLE = value, etc.
2. The format for cards 2 through n, were n represents the number of cards needed to input a case, is starting in column 2 VARIABLE = value, etc.
3. The end of a case is indicated by an ampersand & END following the last data item.

For more information concerning the use of NAMELIST for inputting data consult any S/360 Level H Fortran IV Programmer's Manual. A complete listing of all user inputs can be found in section 14.3 of this book. This listing also indicates whether an input is an integer variable which does not require a decimal point or a real (floating point) variable which does require a decimal point.

4.2 Data Input - Cost and Sizing Models - Although the program user does not input data directly into either the Cost or Sizing Models, a brief discussion of the data link between the executive routine and both of these models follows.

4.2.1 Cost Model - Interface - The interface is a preparatory subroutine for the Spacecraft Cost Model inputs. Since the Spacecraft Cost Model has the capability of estimating many different vehicle configurations, the Interface is required for assembling the outputs of the Sizing Model, Inventroy Model, and Executive Logic to assure the proper inputs to the Spacecraft Cost Model. The inputs to the Interface are tested and assembled to be consistent with the study ground rules, user inputs, and established vehicle configurations. For this reason, the Interface is sensitive to, and constrained by the study ground rules, vehicle configurations, and general program definitions.

The Sizing Model presents the weights in the normal format for weight summaries and not the grouping of weights as used by the Cost Model. The Interface collects these values and assembles them as utilized in the Cost Model. Some inputs required by the Cost Model are internal calculations in the Sizing Model and cannot be passed to the Interface. These internal calculations are simply repeated in the Interface to obtain the desired values.

A series of switches and tests are provided to establish the Cost Model inputs consistent with the study ground rules, user inputs and vehicle configuration.

4.2.2 Sizing Model Inputs - The Sizing Model uses twenty-three input values to run a case, all of which are passed to the program through named common. These input numbers are generated by the executive logic. Twenty-one of these numbers are stored in an array called IBASE which is passed through named common RICE. The other two, ICNFIG and IREUSE are passed through named common COST. variable ICONFIG AND IREUSE are used to calculate an initial vehicle length.

In order to avoid inputting a large number of cards per case, the equivalent of 1300 cards is stored on disk or tape. This includes one or more sets of data for each major subroutine. These are as many as eight different data sets available for a given subroutine (data for the power subrouting has eight possible alternatives), or as few as a single data set (data for the environmental control subroutine.) For each configuration, one set of data is chosen from sets available for each of the twenty-one major subroutines. These twenty one sets of data form the block of data which is read to run a given configuration. There are approximately 1800 numbers read from disk per block. Table 4-1 shows the IBASE numbers from the various configurations.

4.3 Data Output Discussion - There are three basic types of reports generated by this system. i.e. Cost Model Summaries, Sizing Model Weights Statements, and various summary reports generated by the main program.

The general flow or reports through the program is given in Figure 4-2.

4.3.1 Summary Reports - There are seven one page summary reports generated by the program. Each report is generated by subroutine REPORT which is called by the sequence CALL REPORT(I) where I represents the number of the desired summary report. The various reports and associated numbers are as follows:

Report 1 - Listing of input data - This report lists all the user input data required for selection of a cargo weight. (Always generated)

Report 2 - List of current operational variations. This report depicts the switch settings of each operational variation currently being used by the program. (Generated once for each set of operational variations input, i.e., NOPS times).

Report 3 - Summary of inventory and refurbishment quantities. This report summarizes a cycle through the cargo weight loop. It primarily lists the and refurbishment requirements for the current cargo weight. (Generated once for each cargo weight pass.)

Report 4 - Golden Rule Summary - This report summarizes the results of the cargo weight optimization loop and lists the last four cargo weights and associated total program costs as well as the upper and lower boundaries of the cargo weight search.

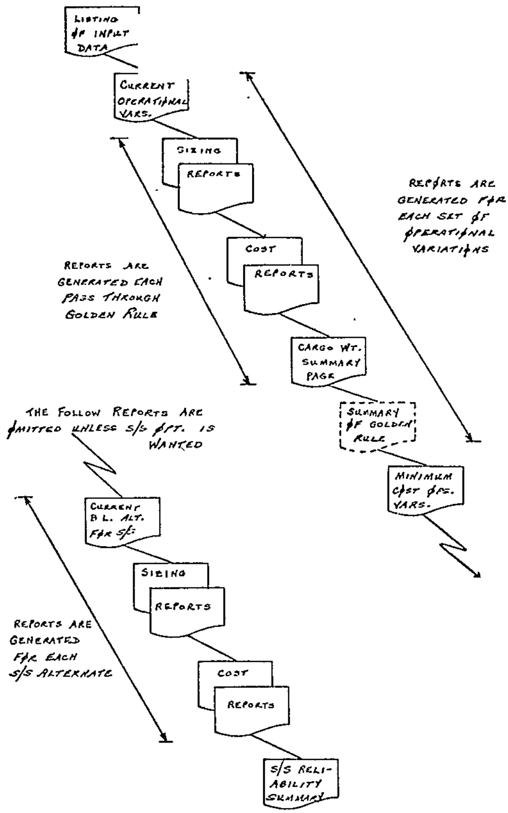
SIZING MODEL BASELINE INDEXES VEHICLE TYPE TABLE 4-1

SUBROUTINE	IA_W	IA_L	IB	IC	ID	IE	IF	IIA	IIB	IIC	IID	IIE	IIF
Personnel	1	1	1	1	1	1	1	1	1	1	1	1	1
Power	1	5	2	3	4	4	4	9	6	7	8	8	8
Propulsion	1	1	1	2	3	4	4	5	5	6	7	8	8
Aero	1	1	1	2	2	2	2	3	3	3	3	3	3
Executive	1	1	1	1	1	1	1	1	1	1	1	1	1
Cargo	2	2	2	1	1	1	1	2	2	1	1	1	1
Landing	2	3	3	2	2	2	2	1	1	1	1	1	1
Miscellaneous	1	1	1	2	2	2	2	1	1	2	2	2	2
Mass Properties	2	2	2	2	2	2	2	1	1	1	1	1	1
Geometry	1	1	1	2	2	2	2	3	3	4	4	4	4
Thermal Protection	1	1	1	1	1	1	1	2	2	2	2	2	2
Size	1	1	1	1	1	1	1	1	1	1	1	1	1
Cost Breakdown	1	1	1	1	1	1	1	1	1	1	1	1	1
Structure	1	1	1	2	2	2	2	3	3	4	4	4	4
Environmental Control	1	1	1	1	1	1	1	1	1	1	1	1	1
Loads	1	1	1	2	2	2	2	3	3	3	3	3	3
Temperature	1	1	1	2	2	2	2	3	3	3	3	3	3
Structural Material	1	1	1	1	1	1	1	1	1	1	1	1	1
Aero Coefficients	1	1	1	1	1	1	1	1	1	1	1	1	1
Propellent Type	1	1	1	1	1	1	1	1	1	1	1	1	1
Engine Configuation	1	1	1	1	1	1	1	1	1	1	1	1	1

This table shows the IBASE values for the various configurations. There are twenty-one values per configuration. Thirteen configurations are shown here.

FIGURE 4-2

GENERAL REPORT FLOW



Report 5 - Minimum Cost Operational Variation - This report lists the operational variation set which resulted in the lowest total program cost.

Report 6 - Summary of Subsystem Optimization. This report indicates the current subsystem baselines being used in the subsystem optimization. It is generated once for each S/S alternate to be considered.

Report 7 - Summary of Subsystem Reliability (Report is only generated if S/S reliability is wanted).

4.3.2 Cost Model Output Description - The following cost summaries are available to the user:

Summary 1 - One page gross cost summary providing spacecraft and launch vehicle cost by program phase.

Summary 2 - Summary 1 expanded to three pages to provide cost by subsystem group and project segment.

Summary 3 - Summary 1 expanded to three pages to provide cost by cost category and program phase.

Summary 4 - This is a complete detailed output of each cost estimating relationship that is programmed for the spacecraft.

Summary 5 - Alphabetical listing of the symbols and input values associated with the estimated spacecraft cost.

Cost summaries 1 and 2 are printed after every pass through the spacecraft cost model. The user then has the option to call for any one or all of the other three summaries. These will also be printed for each pass through the cost model. One switch controls each summary. The user must input these via NAMELIST (see Section 4.1).

PRINT(2)=1 turns on summary 3
PRINT(3)=1 turns on summary 4
PRINT(4)=1 turns on summary 5

To turn these reports off these variables must be set to zero by the user via NAMELIST.

4.3.3 Sizing Model Output - The Sizing Model generates a twenty-eight page report which gives a weight summary of the various subsystems. The detailed weight summaries on pages six through 28 are normally suppressed but can be turned on by making print switch PRNT(1)=1. See Section 4.1 for the method of inputting this number. Table 4-2 contains a list of the reports generated by the Sizing Model.

SIZING MODEL OUTPUT REPORTS

TABLE 4-2

System Description	Page
Table of Contents	1
General Data Summary	2-3
Group Weight Summary	4-5
Geometry Data	6
Mass Characteristics/Mission History	7
Subsystem Data	
Aero Surfaces	8-9
Docking	10
Landing and Recovery	10
Propulsion	11-17
Prime Power	18-19
Power Conversion and Distribution	20
Guidance and Navigation	21
On-Board Checkout	21
Telecommunications	21
Environmental Control System	22
Personnel and Personnel Provisions	26
Crew Station Controls	27
Cargo and Supports	27
Ordinance	27
Ballast	27
Range Safety and Abort	28

5.0 EXECUTIVE OCPDM ROUTINE - This section deals with the executive logic of the OCPDM program which is called MAIN in the actual program. MAIN consists of all the logic which ties the entire program together. In the following section of this book, the details of the significant subroutines are given. This section, however, will deal with the logic and symbols of the MAIN program.

5.1 GENERAL LOGIC - The MAIN program can be divided into three major shares:

a) Phase I
   1) minimum cost operational variation
   2) optimized cargo weight
b) Phase II - Subsystem optimization
c) Phase III - reliability optimization

A general flow diagram illustrating these three phases is given in Figure 5-1.

5.1.1 OPERATIONAL VARIATIONS - The user of the OCPDM program has control over seven major operations activities:

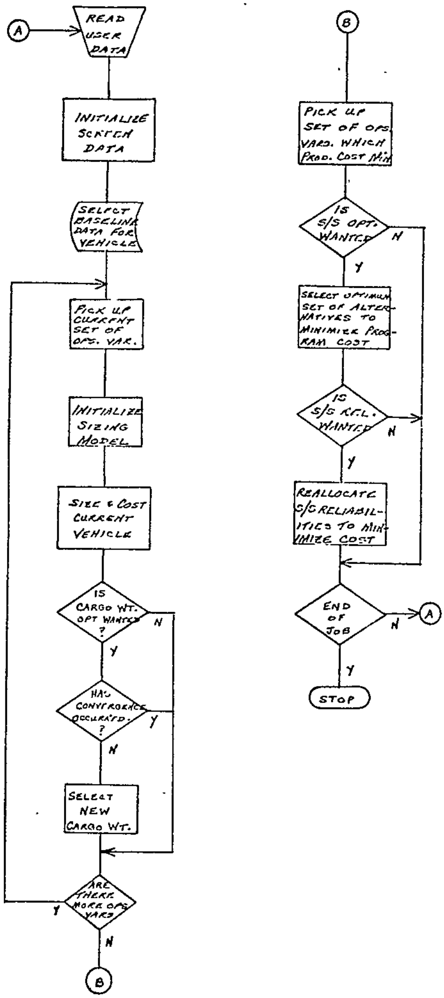
a) launch operations
b) AGE concept
c) refurbishment concept
d) refurbishment site
e) recovery mode
f) recovery site
g) transportation mode

The specific options available for each activity are given in Section 13. The user can input as many "sets" of operational variations as he wishes, but for each set he must specify his choice for each of the seven activities. Specific input instructions are given both in Section 5.4 of this book and in Volume III, Book 3, Section 4. The selection of the optimized operational set is based exclusively on the total program cost. The optimum operational set will be the set of operational activities which produce the lowest program cost. Section 13 of this book discusses the effect of each operations activity.

The operational costs for the cost model are discussed in Volume II, Book 5, Section 6.4.

FIGURE 5-1

MAIN LOGIC FLOW



5.1.2 OPTIMIZED CARGO WEIGHT - In order to evaluate the total program cost for each set of operational variations, the design size and weight of an entire spacecraft must be generated by the spacecraft sizing model (PDAP). Three options are available to the program user:

a) fixed cargo weight/launch
b) fixed total spacecraft weight
c) optimized cargo weight/launch

The user can specify a specific cargo weight/launch and the sizing model will size and weigh an entire spacecraft to accommodate this cargo weight. Alternately, the user can indicate the total spacecraft weight when he is limited to a specific launch vehicle capability. Under this option the PDAP program will find the amount of cargo which the spacecraft can carry. The third alternate is finding the optimized cargo weight, i.e., the cargo weight/launch which will minimize the total program cost for a given set of operational variations. This is really an iterative process with each iteration using a fixed cargo weight. The search technique used to find the minimum cost cargo weight is described in Section 12 of this book. The program user should be aware that each of the three options described in this subsection will be executed for each set of operational activities he inputs into the program.

5.1.3 SUBSYSTEM OPTIMIZATION - After Phase I has been completed, one set of operational activities and either a fixed spacecraft or a fixed cargo weight/launch will be determined and will not be altered during Phases II and III. The subsystem optimization logic examines design alternates for up to 10 different subsystems in the spacecraft. The program user has complete control over which subsystems and which alternates will be examined. There is a set of baseline subsystems which are permanently stored in the program and are held constant throughout Phase I. The specific subsystems which the user can change are:

a) primary structure - entry vehicle
b) primary structure - mission module
c) upper stage boost propulsion
d) orbit maneuver propulsion
e) electrical power
f) hydraulic power
g) guidance and control

h) telecommunications
i) thermal protection
j) environmental control

The subsystem baseline and alternates are described in Section 5.4.3 of this book. The user input instructions are included both in Section 5 of this book and in Volume III, Book 3, Section 4. A detailed descussion of the theory used in the subsystem optimization is given in Section 10 of this book.

5.1.4 RELIABILITY OPTIMIZATION - The final phase of the OCPDM model examines the cost impact of reallocating the reliabilities of the major subsystems. An overall subsystem reliability and upper and lower reliability limits for each subsystem are input by the program user. The optimization logic reallocates the reliabilities of the individual subsystems so as to meet the overall reliability goal at the lowest cost possible. The reallocation process begins by using the subsystem costs generated from Phase II and the reference reliabilities described in Section 6.1.2 of Volume II, Book 1. A detailed description of the reliability optimization model is given in Section 11 of this book.

5.2 SIGNIFICANT VARIABLES - MAIN

ALT(I,J)	-	Jth alternate for the Ith subsystem (subsystem optimization)
ATS	-	Air transportation switch (cost model)
AZ	-	Launch azimuth (degrees)
BAL	-	Ballistic configuration switch (cost model)
BALT (I)	-	Baseline alternate for the Ith subsystem (subsystem optimization)
BTS	-	Water transportation switch (cost model)
C(I)	-	Total program cost in millions of dollars (subsystem optimization)
CØUNT	-	Counts number of iterations for a Golden Rule search
DIFF	-	Difference in years between 1969 and the dollar base year
E2S	-	2 existing recovery sites indicator (cost model)
EXX	-	Learning exponent for the recertification time (inventory model)
G	-	Golden Rule constant $G = 2/(1+\sqrt{5}) \cong .618$

GØLD	-	Golden Rule indicator = 1 on = 0 off
HFT	-	Hot fire acceptance test (cost model)
IBV	-	Ballistic configuration switch for REUSE = D, E or F (cost model)
ILB	-	Lifting body configuration switch for REUSE = D, E or F (cost model)
INF	-	Inflation factor
ISP	-	Specific impulse of the propellant in feet/second (subsystem optimization)
L1M1	-	Number of iterations for the Golden Rule search
LLM	-	Land landing mode switch (cost model)
LNG	-	Natural logarithm of the Golden Rule constant
LØØP	-	Upper limit on the cargo weight/launch optimization DØ loop
LØW	-	Lower limit on the cargo weight/launch optimization DØ loop
LSPH1	-	Latitude in degrees of the launch site
LTS	-	Land transportation switch (cost model)
LVTW	-	Launch vehicle throw weight capability in pounds for the input orbit inclination and launch site
MBV	-	Ballistic configuration switch for REUSE = A, B. or C (cost model)
MINC	-	Total program cost in millions of dollars for the minimum cost operational variations set
MINC(I)	-	Total program cost in millions of dollars for the minimum cost alternate of the Ith subsystem
MLB	-	Lifting body configuration for REUSE = A, B, or C
MØV	-	Operational variations set number for the minimum total program cost
NISP	-	Specific impulse of the propellant in feet/second (subsystem optimization)
NØDL	-	Number of launches during the development phase
NRF	-	Refurbishment site indicator (cost.model)
NS	-	Number of recovery sites (cost model)
ØCWL	-	Optimized cargo weight/launch for each operational set
ØMB	-	Orbit maneuver baseline alternate (subsystem optimization)
P	-	Degradation factor for the launch vehicle throw weight capability

PC(I,J) - Total program cost in millions of dollars for the Jth alternate of the Ith subsystem (subsystem optimization)

R(I) - Slope of the program cost versus orbit maneuver or upper stage boost weight (subsystem optimization)

SCWT - Total spacecraft weight in pounds

a. if REUSE = A, B, or C
SCWT = effective spacecraft weight

b. if REUSE = D, E. or F
SCWT = gross spacecraft weight

SMALL - Total program cost for the optimized cargo weight/launch for each operational variations set

SPAN - Difference in pounds between the initial upper and lower limits of the Golden Rule search range

SS - Subsystem indicator (subsystem optimization)

START - Lower limit in pounds of the search range for each iteration of the Golden Rule search

TDS - Test deletion switch (cost model)

TPC(I) - Total program cost in millions of dollars for the four current values used in the Golden Rule search

USP - Integral upper stage propulsion switch (cost model)

VLM - Vertical landing mode switch (cost model)

WT(I) - Four current cargo weight/launch used in the Golden Rule search

WTØM (I,J)- Weight of the orbit maneuver system in pounds for the Jth alternate of the Ith subsystem (subsystem optimization)

WTUS(I,J) - Weight of the upper stage propulsion in pounds for the Jth alternate of the Ith subsystem (subsystem optimization)

5.3 PROGRAM USER INPUT VARIABLES

5.3.1 CONFIGURATION DATA - Two variable inputs determine the configuration type and the reuse category of the entry vehicle. The configuration and reuse are two of the most important variables for the entire program. The two user inputs are:

CØNFIG - Configuration type indicator
= 1 ballistic
= 2 M2/F2 lifting body

REUSE - Reuse category indicator
= 1 all expendable vehicle with separate mission module
= 2 reusable entry vehicle with separate mission module
= 3 reusable entry vehicle with integral cargo section
= 4 reusable entry vehicle with integral cargo and upper stage propulsion with expendable tip tanks
= 5 reusable entry vehicle with integral cargo, upper stage propulsion and tankage
= 6 same entry vehicle as REUSE=5 except a reusable first stage is used.

5.3.2 GENERAL DATA -

CD - Cargo Density (lbs/ft^3)

CS - Crew Size - number of men

DL - Design life of the reusable entry vehicle (number of flights

INC - Orbit inclination in degrees (0° - 90°)

LS - Launch site
= 1 ETR (28.5°)
= 2 WTR (34.0°)

LVT - Launch vehicle type
a. expendable first and second stages
= 1 solid/liquid
= 2 liquid/liquid
b. expendable first stage only
= 3 solid
= 4 liquid
c. reusable first stage
= 5 VTOHL liquid

Note: If LVT = 0, program will
(1) compare LV's 1 and 2 if REUSE = 1, 2, or 3
(2) compare LV's 3 and 4 if REUSE = 4 or 5
(3) only use LVT = 5 if REUSE = 6

ØST - Orbit stay time (docked to space station) in days

PHDELV - Phasing ΔV in feet/second

PØMS(1) - Mission success probability for the launch vehicle

PØMS(2) - Mission success probability for the spacecraft

PØSR - Launch to launch reliability for the entry vehicle (if vehicle is expendable, define PØSR as launch to recovery reliability)

STGVEL - Staging velocity of the first stage of the launch vehicle in feet/second (solid first stage launch vehicle only) - program adjusts the nominal ΔV of the upper stage by an amount equal to 10,600 - STGVEL. (see Volume II, Book 1, section 4 and Book 5, section 7).

5.3.3 OPERATIONAL VARIATIONS INPUT DATA -

NØPS - Number of sets of operational variations
Note: NØPS must equal at least 1 (there are 7 operations indicator for each set)

AGE - AGE indicator
= 1 semiautomatic
= 2 on-board checkout

LØ - Launch operations concept indicator
= 1 Gemini style
= 2 integrated checkout

RECS - Recovery site indicator
= 1 2 new sites
= 2 2 existing sites
= 3 3 existing sites
= 4 4 existing sites

RECM - Recovery mode indicator
= 1 water
= 2 land

REFP - Refurbishment concept indicator
= 1 scheduled maintenance (with hot firing tests)
= 2 scheduled maintenance (no hot firing tests)
= 3 limited maintenance

LØF - Launch operations factor

REFS - Refurbishment sites
= 1 factory
= 2 new site

TRANS - Transportation mode
= 1 water
= 2 land
= 3 air

5.3.4 SIZING MODEL INPUT DATA -

CWL - Cargo weight/launch in pounds

FR - Final range of the Golden Rule search in pounds

LL - Lower limit of the initial search range in pounds

LR - Annual launch rate

LVTWE - Launch vehicle throw weight capability in pounds for a due east launch from ETR

PL - Operational program length in years

TCW - Total cargo weight in pounds delivered to orbit for the entire program

UL - Upper limit of the initial search range in pounds

5.3.5 COST MODEL INPUT DATA -

CB - Dollar base year

IR - Annual inflation factor expressed in decimal form

KENGR - Labor rate ($/hr) for engineering

KLRS - Labor rate ($/hr) remote site

KPRØD - Labor rate ($/hr) for production

KTØØL - Labor rate ($/hr) for tooling

5.3.6 SUBSYSTEM OPTIMIZATION INPUT DATA -

NALT(I) - Number of alternates to be examined for each subsystem
I = 1 primary structure entry vehicle
2 primary structure mission
3 upper stage boost propulsion
4 orbit maneuver propulsion
5 electrical power
6 hydraulic power
7 guidance and control

8 telecommunications
9 thermal protection
10 environmental control

STEV(I) - Alternates for the primary structure for the entry vehicle

STMM(I) - Alternates for the primary structure for the mission module

USB(I) - Alternates for the upper stage boost propulsion

ØM(I) - Alternates for the orbit maneuver propulsion

ELECT(I) - Alternates for the electrical power system

HYD(I) - Alternates for the hydraulic power system

GC(I) - Alternates for the guidance and control systems

TELE(I) - Alternates for the telecommunications system

TPS(I) - Alternates for the thermal protection system

ECS(I) - Alternates for the environmental control system

5.3.7 RELIABILITY OPTIMIZATION INPUT DATA -

SSREL - Total subsystem reliability

ECSR(I) - Upper (I=2) and lower (I=1) reliability limits for environmental control

GCR(I) - Upper and lower reliability limits for guidance and control

ØACR(I) - Upper and lower reliability limits for orbit attitude control

PØWR(I) - Upper and lower reliability limits for electrical and hydraulic power system

RACR(I) - Upper and lower reliability limits for reentry attitude control

TELER(I) - Upper and lower reliability limits for the telecommunications system

TPSR(I) - Upper and lower reliability limits for the thermal protection system

USBR(I) - Upper and lower reliability limits for the upper stage boost propulsion

VERNR(I) - Upper and lower reliability limits for the vernier maneuver system

5.3.8 PRINT CONTROL INPUTS -

PRINT(1) - Control switch on the sizing model printout
= 0 abbreviated 7 page summary of the spacecraft weights
= 1 expanded 28 page summary showing subsystem by subsystem weights

PRINT(2) - Control switch on summary 3 of the cost model output
= 0 no printout
= 1 three page cost summary showing costs by cost category (e.g., engineering, tooling, etc.) and program phase

PRINT(3) - Control switch on summary 4 of the cost model output
= 0 no printout
= 1 19 page cost summary showing the results of every CER

PRINT(4) - Control switch on summary 5 of the cost model output
= 0 no printout
= 1 alphabetical listing of the input values to the cost model

NOTE: The cost summaries are discussed in Section 7 of this book.

5.4 INPUT DATA LOGIC

5.4.1 OPERATIONAL VARIATIONS - The number of sets of operational variations is input as NØPS. NØPS must be a positive integer and tells the program to read in and examine NØPS sets. Each set consists of 7 operations indicators:

a) launch operations concept (LØ)
b) AGE concept (AGE)
c) recovery mode (RECM)
d) recovery site (RECS)
e) refurbishment concept (REFP)
f) refurbishment site (REFS)
g) transportation mode (TRANS)

The sets may differ in all 7 operations categories, in only 1 category, or not at all (this last case would yield two identical iterations from the program)

The launch operations factor (LOF) is a judgement factor (usually less than unity) applied to industrial area, on-pad testing, and quality assurance activities and determined by comparing the man-hour expenditure of the postulated launch operations to Gemini launch operations.

There are some limitations on the operations indicators depending on the vehicle reuse category. The water recovery mode (RECM = 1) can only be used for a IA vehicle, i.e., an all expendable ballistic. For all other configurations a land recovery is used internally and if the user puts RECM = 1, the program will change it to RECM = 2. Also, for a water landing IA vehicle, the recovery site indicator (RECS) is not read and does not then affect the program logic. For the all expendable vehicles, the refurbishment concept (REFP) and refurbishment site (REFS) indicators are not read and do not affect the program logic.

5.4.2 CARGO WEIGHT OPTIMIZATION DATA INPUTS - There are three basic modes which the program user has available when sizing the spacecraft:

a) fixed cargo weight/launch
b) fixed spacecraft weight
c) optimized cargo weight/launch obtained through a Golden Rule search.

Tables 5-1, 5-2, and 5-3 summarize the data input requirements for each mode. These tables only show the critical input variables and the user is still required to input appropriate values for all the other input variables. The five input variables which determine which mode the program will run are:

a) TCW - total cargo weight in pounds delivered to orbit
b) CWL - cargo weight/launch in pounds
c) PL - operational program length in years
d) LR - annual launch rate
e) LVTWE - launch vehicle throw weight capability for a due east ETR launch.

For all three modes one or two variables may be input as zero to indicate a particular option. In every case, the program calculates a value for these variables and prints it out.

When a fixed spacecraft weight is desired, a value is input for LVTWE as indicated in Table 5-2. This launch vehicle throw weight capability is adjusted internally to account for 50°, 70°, or 90° orbit inclinations and a WTR or ETR launch site. The adjustment factors are shown in Table 5-4.

TABLE 5-1

FIXED CARGO WEIGHT/LAUNCH DATA INPUTS

LVTWE = 0	
CWL ≠ 0	
TCW PL LR	Set any one to zero and input nonzero values for the remaining two

NOTE: If nonzero values for CWL, TCW, PL and LR are input, then

TCW = (CWL) x (LR) x (LR)

must be true and the mission success reliability (POMS (1) x POMS (2)) must equal unity.

TABLE 5-2

FIXED SPACECRAFT WEIGHT DATA INPUT

LVTWE ≠ 0	
CWL = 0	
TCW PL LR	Set any one to zero and input nonzero values for the remaining two

NOTE: LVT ≠ 0

TABLE 5-3

GOLDEN RULE SEARCH FOR OPTIMIZED CARGO WEIGHT/LAUNCH

DATA INPUTS

LVTWE = 0	
CWL = 0	
TCW LR PL	Set any one to zero and Input nonzero values for the remaining two

TABLE 5-4

TRUE LAUNCH VEHICLE THROW WEIGHT

Launch vehicle throw weight capability is input for a due east launch from ETR (LVTWE)

The true launch vehicle throw weight capability (LVTW) for different orbit inclinations and launch sites is derived from:

$$LVTW=(P)^{*}x(LVTWE), 0 \leq P \leq 1$$

Orbit Inc/Launch Site	P
28.5°/ETR	1.0
50°/ETR	.9488
70°/ETR	.8874
90°/WTR	.8223

As indicated above the optimized cargo weight/launch is found with a search technique known as the Golden Rule. The theory and use of this technique is described in Section 12 of this book. There are three additional data inputs for the search mode. The program user must indicate the upper and lower limits of the search range, i.e., UL and LL, in pounds. The final accuracy of the optimized cargo weight/launch is controlled by indicating the final range of the optimized weight in pounds. This means that if 1000 pounds is input for the final range, FR, the search technique will iterate until the search range is less than or equal to 1000 pounds, i.e., the optimized cargo weight is within 1000 pounds of the true optimal cargo weight.

5.4.3 SUBSYSTEM OPTIMIZATION - The subsystem optimization phase allows the user to examine possible alternates for 10 different subsystems. Tables 5-5 through 5-13 list a description of each alternate for all 10 subsystems and also indicates for which configurations the alternates are applicable. The number of alternates possible for a subsystem varies from subsystem to subsystem. Structure for example, has 12 alternates while orbit maneuver has 8 alternates and hydraulic power has only 2. The tables also indicate which baseline alternate is used for each vehicle configuration.

The subsystem optimization logic expects data for each subsystem to be input in two locations:

a) Input the number of alternates to be examined for each subsystem in the NALT array. The NALT array contains 10 slots as identified in Section 5.3.4 of this book. If NALT(1) = 0, then the program assumes no alternates will be run against the baseline for the entry vehicle primary structure. If, however, NALT(5) = 1, this indicates there is one alternate to be compared with the electrical power baseline. If the entire NALT array is zeroed out, the subsystem optimization logic is bypassed.

b) If a NALT slot indicates there are some alternates to be compared with the baseline alternate, then the corresponding array for that subsystem will be read to find out which alternates are to be used. For example, if NALT(5) = 1, then one alternate is expected in the

TABLE 5-5

PRIMARY STRUCTURE ALTERNATES

ENTRY VEHICLE

Alternate Designation	Concept Definition: Methods of Construction	Concept Definition: Material	Vehicle Application
STEV = 1	Single-skin with frames	Aluminum	All
STEV = 2*	Sheet-stringer with frames	Aluminum	↑
STEV = 3	Single-skin, corrugations with frames	Aluminum	
STEV = 4	Single-skin with frames	Magnesium	
STEV = 5	Sheet-stringer with frames	Magnesium	
STEV = 6	Single-skin, corrugation with frames	Magnesium	
STEV = 7	Single-skin with frames	Titanium	
STEV = 8	Sheet-stringer with frames	Titanium	
STEV = 9	Single-skin, corrugations with frames	Titanium	
STEV = 10	Single-skin with frames	Steel	
STEV = 11	Sheet-stringer with frames	Steel	↓
STEV = 12	Single-skin, corrugations with frames	Steel	All

* Baseline Concept

TABLE 5-6

Primary Structure Alternates

Mission Module

Alternate Designation	Concept Definition: Methods of Construction	Concept Definition: Material	Vehicle Application
STMM = 1	Single-skin with frames	Aluminum	IA, B IIA, B
STMM = *2	Sheet-stringer with frames	Aluminum	
STMM = 3	Single-skin, corrugations with frames	Aluminum	
STMM = 4	Single-skin with frames	Magnesium	
STMM = 5	Sheet-stringer with frames	Magnesium	
STMM = 6	Single-skin, corrugation with frames	Magnesium	
STMM = 7	Single-skin with frames	Titanium	
STMM = 8	Sheet-stringer with frames	Titanium	
STMM = 9	Single-skin, corrugations with frames	Titanium	
STMM = 10	Single-skin with frames	Steel	
STMM = 11	Sheet-stringer with frames	Steel	
STMM = 12	Single-skin, corrugations with frames	Steel	IA, B IIA, B

* Baseline Concept

TABLE 5-7

Alternate Summary - Upper Stage Boost Propulsion System

Alternate No.	Propellant	Engine Design	Vehicle Applications
USB = 1*	O_2/H_2	Bell	
USB = 3	F_2/H_2	Bell	I/D, E, F
USB = 4	Flox/CH_4	Bell	II/D, E, F
USB = 5	NTO/A-50	Bell	

* Baseline System Concept

TABLE 5-8

Alternate Summary - Orbital Maneuver

Alternate No.	System Function	Propellant	Engine Groups	Vehicle Application
OM = 1	A - All functions	NTO/MMH	Single	I/A, B
OM = 2*	A - All functions	NTO/MMH	Dual	All
OM = 3	A - Docking/AC	NTO/MMH	Single	I/A, B
	B - Ascent/Phasing/Deorbit	O_2/H_2	Single	II/A, B
OM = 4	A - Docking/AC	NTO/MMH	Dual	All
	B - Descent/Phasing/Deorbit	O_2/H_2	Single	
OM = 5**	A - Ascent/Docking/Phasing/AC	NTO/MMH	Single	I/A, B
	B - Deorbit	Solid	Single	II/A, B
OM = 6	A - Ascent/Docking/Phasing/AC	NTO/MMH	Dual	All
	B - Deorbit	Solid	Single	
OM = 7	A - Docking/AC	NTO/MMH	Single	I/A, B
	B - Ascent/Phasing	O_2/H_2	Single	II/A, B
	C - Deorbit	Solid	Single	
OM = 8	A - Docking/AC	NTO/MMH	Dual	All
	B - Ascent/Phasing	O_2/H_2	Single	
	C - Deorbit	Solid	Single	

AC = Attitude Control

* Baseline concept selection - Integral vehicles (C, D, E, F)

** Baseline concept selection - Modular vehicles (A, B)

TABLE 5-9

Power Subsystem Alternates

Alternate No.	Ascent and Phasing	Re-entry	Vehicle Application
ELECT = 1*	Batteries in MM	Batteries in EV	IA, IIA
ELECT = 2*	Fuel Cells and Batteries in EV, Reactant Supply in MM	Batteries in EV	IB, IIB
ELECT = 3*	Fuel Cells, Batteries and Reactant Supply in EV	Batteries in EV	IC thru IF IIC thru IIF
ELECT = 4	Batteries in EV	Batteries in EV	1A (Land Landing) 1B thru IF IIA thru IIF
ELECT = 5	Fuel Cells and Reactant in MM	Batteries in EV	IA, IB IIA, IIB

Hydraulic Power

HYD = 1	Batteries	IIA, IIB
HYD = 2*	Turbine	IIA thru IIF

* Baseline Concept

TABLE 5-10

Guidance and Control Subsystem Alternates

Alternate No.	Concept Definition	Vehicle Application
GC = 1*	Single IMU, Computer, FCS	IA (Water Landing)
GC = 2*	Gimballed and Strap-down IMU Dual Computers, Single FCS	IC
GC = 3*	Gimballed and Strap-down IMU Dual Computers, Redundant FCS	IA (Land Landing), IB IIA, IIB and IIC
GC = 4*	2 Gimballed and 1 Strap-down IMU, Dual Computers, Redundant FCS	ID, IE, IF, IID, IIE, IIF
GC = 5**	Advanced Concept of GC-1	IA (Water Landing)
GC = 6**	Advanced Concept of GC-2	IC
GC = 7**	Advanced Concept of GC-3	IA (Land Landing), IB, IIA, IIB and IIC
GC = 8**	Advanced Concept of GC-4	ID, IE, IF, IID, IIE, IIF

** Advanced concepts employ advanced circuit techniques and replace gimballed platforms with strap-down platforms.

TABLE 5-11

Telecommunication Systems Alternates

Alternate No.	System Definition	Application (Vehicle Configuration)
TELE = 1	Separate Systems (Gemini Type Concept	All
TELE = 2*	Unified S-Band System (Apollo Type Concept)	All
TELE = 3	Unified S-Band System with Integral Rendezvous Ranging Capability	All
TELE = 4	Same as TC-2 but with the use of Advanced Circuit Techniques	All
TELE = 5	Same as TC-3 but with the use of Advanced Circuit Techniques	All

* Baseline Concept

TABLE 5-12

Thermal Protection System Alternates

Alternate No.	Maximum Allowable Temperature for Rad. Shingles – °F	Inner Shell Cooling	Reuse	Vehicle Application
TPS = 1	400	Passive	Yes	All
TPS = 2	1200	Passive	Yes	All
TPS = 3	1600	Passive	Yes	All
TPS = 4	2400	Passive	Yes	All
TPS = 5*	3100	Passive	Yes	All
TPS = 6	3500	Passive	No	All
TPS = 7	400	Passive/Active	Yes	All
TPS = 8	1200	Passive/Active	Yes	All
TPS = 9	1600	Passive/Active	Yes	All
TPS = 10	2400	Passive/Active	Yes	All
TPS = 11	3100	Passive/Active	Yes	All
TPS = 12	3500	Passive/Active	No	All

* Baseline Concept

TABLE 5-13

ECLS Subsystem Alternates

Alternate No.	Subsystem Definition	Vehicle Application
ECS = 1*	5 psi O_2 atmosphere - gaseous storage Water boiler and LiOH in EV Primary O_2, H_2O and radiator in MM	IA, IB, IIA, IIB
ECS = 2*	5 psi O_2 atmosphere - gaseous storage Water boiler and LiOH	IC, ID, IE, IF IIC, IID, IIE, IIF
ECS = 3	5 psi O_2 atmosphere - cryo storage Water boiler and LiOH	IC, ID, IE, IF IIC, IID, IIE, IIF
ECS = 4	5 psi O_2 atmosphere - gaseous storage Water boiler and LiOH in EV Primary O2 and H_2O in MM	IA, IB, IIA, IIB
ECS = 5	5 psi O_2 atmosphere - cryo storage Water boiler and LiOH in EV Primary O2 and H_2O in MM	IA, IB, IIA, IIB
ECS = 6	5 psi O_2 atmosphere - cryo storage Water boiler, LiOH and primary O_2 in EV H_2O in MM	IB, IIB
ECS = 7	5 psi O_2 atmosphere - cryo storage Water boiler and LiOH in EV Primary O2, H_2O and radiator in MM	IA, IB, IIA, IIB
ECS = 8	5 psi O_2 atmosphere - cryo storage Water boiler, LiOH and primary O_2 in EV, Radiator and H_2O in MM	IB, IIB

* Baseline subsystem

electrical power array, in this case, ELECT. If the program reads ELECT(1) = 4, it will use the number 4 alternate for electrical power, i.e. batteries in the entry vehicle, and compare the cost with the baseline electrical alternate number 1 which uses batteries both in the entry vehicle and the mission module.

Tables 5-14 through 5-22 list each vehicle configuration and indicate all the alternates for each subsystem which are applicable for that configuration.

5.4.4 <u>SUBSYSTEM RELIABILITY OPTIMIZATION DATA INPUTS</u> - The definitions for the reliability optimization model are given in Section 5.3.7 of this book. Since all the inputs are reliabilities there are restrictions on these values; obviously, the reliabilities must be less than unity and greater than zero.

The program user must input one overall reliability goal and an upper and lower reliability limit for each of the subsystems. The reliability of a particular subsystem can be fixed by setting both the upper and lower limit equal to the desired reliability. Before the reliability optimization begins, the product of the lower limit reliabilities is calculated and if this product is greater than or equal to the reliability goal, the program uses the lower limit reliability for each subsystem. The product of the upper limits is also calculated and if the product is less than or equal to the reliability goal, the program uses the upper limit reliability for each subsystem.

A discussion of the theory and use of the reliability optimization model is given in Section 11 of this book.

TABLE 5-14

Subsystem Alternates for a IA (Water Landing) Spacecraft

Subsystems	Applicable Alternates
1. Primary Structure Entry Vehicle	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
2. Primary Structure Mission Module	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
3. Upper Stage Propulsion	N.A.
4. Orbit Maneuver Propulsion	1, 2, 3, 4, 5, 6, 7, 8, 5*
5. Electrical Power	1*, 5
6. Hydraulic Power	N.A.
7. Guidance and Control	1*, 5
8. Telecommunications	1, 2, 3, 4, 5, 2*
9. Thermal Protection	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 5*
10. Environmental Control	1*, 4, 5, 7

* Baseline alternate for this vehicle

TABLE 5-15

Subsystem Alternates for a IA (Land Landing) Spacecraft

Subsystems	Applicable Alternates
1. Primary Structure Entry Vehicle	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
2. Primary Structure Mission Module	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
3. Upper Stage Propulsion	N.A.
4. Orbit Maneuver Propulsion	1, 2, 3, 4, 5, 6, 7, 8, 5*
5. Electrical Power	1*, 4, 5
6. Hydraulic Power	N.A.
7. Guidance and Control	3*, 7
8. Telecommunications	1, 2, 3, 4, 5, 2*
9. Thermal Protection	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 5*
10. Environmental Control	1*, 4, 5, 7

* Baseline alternate for this vehicle.

TABLE 5-16

Subsystem Alternates for a IB Spacecraft

Subsystems	Applicable Alternates
1. Primary Structure Entry Vehicle	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
2. Primary Structure Mission Module	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
3. Upper Stage Propulsion	N.A.
4. Orbit Maneuver Propulsion	1, 2, 3, 4, 5, 6, 7, 8, 5*
5. Electrical Power	2*, 4, 5
6. Hydraulic Power	N.A.
7. Guidance and Control	3*, 7
8. Telecommunications	1, 2, 3, 4, 5, 2*
9. Thermal Protection	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 5*
10. Environmental Control	1*, 4, 5, 6, 7, 8

* Baseline alternate for this vehicle.

TABLE 5-17

Subsystem Alternates for a IC Spacecraft

Subsystems	Applicable Alternates
1. Primary Structure Entry Vehicle	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
2. Primary Structure Mission Module	N.A.
3. Upper Stage Propulsion	N.A.
4. Orbit Maneuver Propulsion	2*, 4, 6, 8
5. Electrical Power	3*, 4
6. Hydraulic Power	N.A.
7. Guidance and Control	2*, 6
8. Telecommunications	1, 2, 3, 4, 5, 2*
9. Thermal Protection	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 5*
10. Environmental Control	2*, 3

* Baseline alternate for this vehicle.

TABLE 5-18

Subsystem Alternates for a ID or IE Spacecraft

Subsystems	Applicable Alternates
1. Primary Structure Entry Vehicle	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
2. Primary Structure Mission Module	N.A.
3. Upper Stage Propulsion	1, 2, 3, 4, 1*
4. Orbit Maneuver Propulsion	2*, 4, 6, 8
5. Electrical Power	3*, 4
6. Hydraulic Power	N.A.
7. Guidance and Control	4*, 8
8. Telecommunications	1, 2, 3, 4, 5, 2*
9. Thermal Protection	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 5*
10. Environmental Control	2*, 3

* Baseline alternate for this vehicle.

TABLE 5-19

Subsystem Alternates for a IIA Spacecraft

Subsystems	Applicable Alternates
1. Primary Structure Entry Vehicle	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
2. Primary Structure Mission Module	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
3. Upper Stage Propulsion	N.A.
4. Orbit Maneuver Propulsion	2, 3, 4, 5, 6, 7, 8, 5*
5. Electrical Power	1*, 4, 5
6. Hydraulic Power	1, 2*
7. Guidance and Control	3*, 7
8. Telecommunications	1, 2, 3, 4, 5, 2*
9. Thermal Protection	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 5*
10. Environmental Control	1*, 4, 5, 7

* Baseline alternate for this vehicle.

TABLE 5-20

Subsystem Alternates for a IIB Spacecraft

Subsystems	Applicable Alternates
1. Primary Structure Entry Vehicle	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
2. Primary Structure Mission Module	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
3. Upper Stage Propulsion	N.A.
4. Orbit Maneuver Propulsion	2, 3, 4, 5, 6, 7, 8, 5*
5. Electrical Power	2*, 4, 5
6. Hydraulic Power	1, 2*
7. Guidance and Control	3*, 7
8. Telecommunications	1, 2, 3, 4, 5, 2*
9. Thermal Protection	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 5*
10. Environmental Control	1*, 4, 5, 6, 7, 8

* Baseline alternate for this vehicle.

TABLE 5-21

Subsystem Alternates for a IIC Spacecraft

Subsystems	Applicable Alternates
1. Primary Structure Entry Vehicle	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
2. Primary Structure Mission Module	N.A.
3. Upper Stage Propulsion	N.A.
4. Orbit Maneuver Propulsion	2*, 4, 6, 8
5. Electrical Power	3*, 4
6. Hydraulic Power	2*
7. Guidance and Control	3*, 7
8. Telecommunications	1, 2, 3, 4, 5, 2*
9. Thermal Protection	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 5*
10. Environmental Control	2*, 3

* Baseline alternate for this vehicle.

TABLE 5-22

Subsystem Alternates for a IID, IIE or IIF Spacecraft

Subsystems	Applicable Alternates
1. Primary Structure Entry Vehicle	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 2*
2. Primary Structure Mission Module	N.A.
3. Upper Stage Propulsion	1, 2, 3, 4, 1*
4. Orbit Maneuver Propulsion	2*, 4, 6, 8
5. Electrical Power	3*, 4
6. Hydraulic Power	2*
7. Guidance and Control	4*, 8
8. Telecommunications	1, 2, 3, 4, 5, 2*
9. Thermal Protection	1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 5*
10. Environmental Control	2*, 3

* Baseline alternate for this vehicle.

SIZING MODULE DOCUMENTATION

6.0 SIZING MODULE DOCUMENTATION - The sizing module in the program serves as an interface between Mission Requirements and the Design Criteria (input to the program) and the final costing of the concept (output of the program). The sizing module converts the input into the physical parameters of weight, size (length, volume, areas, etc.), and definition of all subsystem design characteristics, to be used as partial input to the costing analysis. This section provides the engineering documentation for the sizing module. Figure 6.0-1 is a table of contents showing the order in which the many subroutines are discussed.

6.1 Executive Program - The executive program for the Sizing Module is subroutine PDAP within the larger sizing analysis program. Subroutine PDAP directs the flow of computations through sixteen primary subroutines and tests for convergence of the results. The end product, after convergence, is a vehicle sized and weighed to meet a set of input requirements and mission criteria. The logic flow is shown in Figure 6.1-1.

The first routine is the personnel (PRSNL) subroutine which defines weight for the personnel and their provisions.

The geometry (GEOM) routine provides all geometric characteristics for the vehicle including thirteen body panel wetted areas, five increments of inner moldline volumes, perimeters, equivalent radii, and geometric centers at five body cuts. The aerodynamic surface areas and their geometric centers are also defined. The routine operates from a set of special coordinates which are developed by a lofting program.

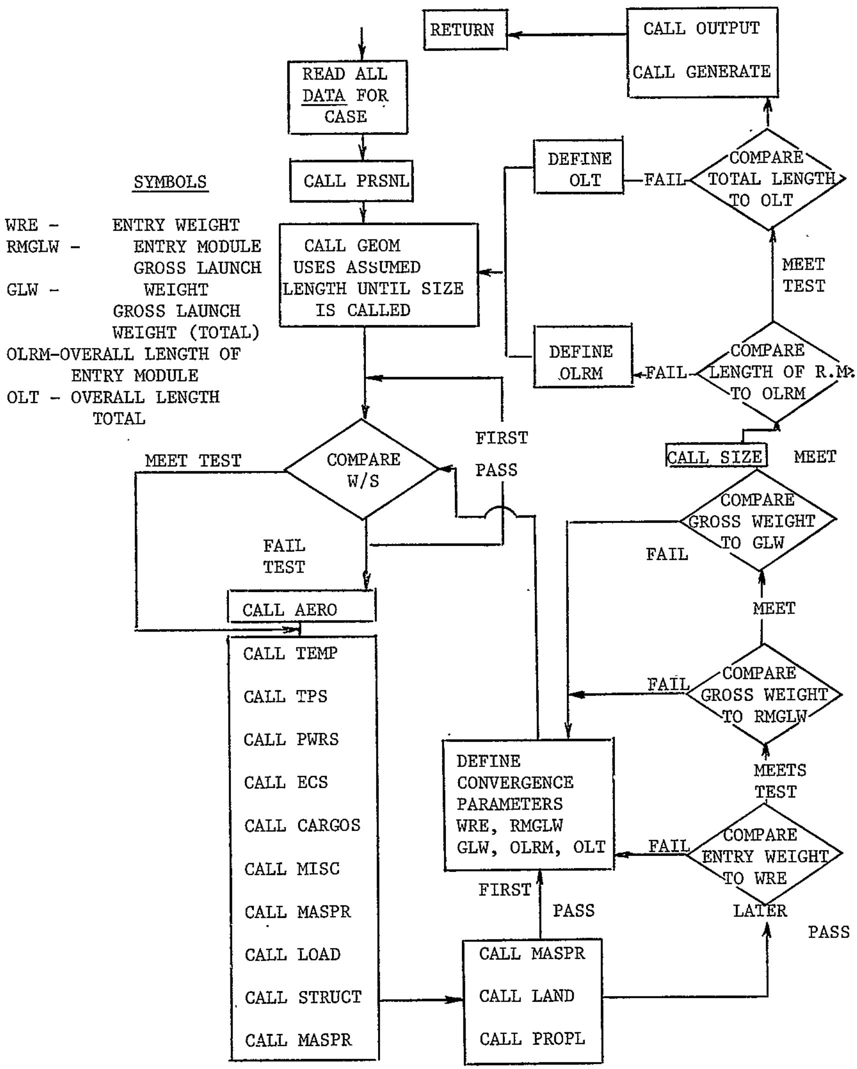
After completing the geometry routine, the entry trajectory data is calculated. The closed form trajectory analysis of the aerodynamic (AERO) routine computes velocity, altitude and density versus time for two entry modes - cross range and minimum down range. Next the temperature (TEMP) routine is called to calculate the temperature, heating rates, total heat, and heating time for twenty-seven (27) points on the ship. This routine uses the output of the AERO routine as input to make the thermodynamic computations. The temperature routine output is design data for the thermal protection (TPS) routine as the maximum temperature and maximum total heat for each section is stored

FIGURE 6.0-1

ENGINEERING DOCUMENTATION OF THE SIZING MODULE

FIGURE 6.1-1

SIZING MODEL SIMPLIFIED LOGIC DIAGRAM



along with the corresponding heating time and heating rate. This data provides a design envelope for the weighing of the thermal protection system.

Next the power (PWRS) subroutine is called to weigh the power generation equipment-batteries or fuel cells - the power distribution system, and the aerodynamic control surface power system. The Environmental Control System (ECS) subroutine computes weight items which maintain the internal heat balance of the spacecraft and which provide a habitable environment for man. The cargo (CARGOS) subroutine computes the distribution of launch and return cargo between the entry module and the mission module given either the total cargo at launch or the effective launch weight limit. The miscellaneous (MISC) subroutine picks up such weight items as recovery, docking, flotation, communication, on-board checkout and instrumentation.

The mass properties (MASPR) subroutine computes the total spacecraft weight at eight points in the mission. In addition, the vehicle is balanced at the entry condition to meet the performance required center of gravity. Inertias, products of inertia, and center of gravities are computed at all mission points.

The load (LOAD) subroutine computes launch and landing body bending moments, axial loads and shears using the aerodynamic load factor $C_{N\alpha}$, and C_p on four arbitrary sections on the body. The maximum launch αq, dynamic pressure q, aeroelastic and buffet factors are additional inputs used in conjunction with the mass distribution generated in the mass properties routine. These loads are then passed to the structures (STRUCT) subroutine which computes the inner body shell and ring weight using an analytical model which includes hardware data correlation factors. Both shell and ring analysis is a function not only of the loading conditions for the particular body shape but also a function of the body geometry. This data is computed in the geometry routine and passed to the structure subroutine. The structures subroutine also computes additional structural penalties which exist in the vehicle flooring, windows, hatches, bulkheads and the structural portion of the aerodynamic surfaces.

The propulsion (PROPL) subroutine computes all propellant quantities required to meet the input performance parameters - incremental velocity between mission points and impulse divided by mission point weight. Then thrust levels are computed, engine weights, tankage for the propellant and miscellaneous weight to complete the total propulsion system weight. Fourteen separate systems are available ranging from an integral boost system to orbit attitude control and entry control systems. The propulsion routine uses output of the mass properties routine as input to define mission point weights. The last subroutine called is the landing (LAND) subroutine which computes landing gear weights and chute weights.

The first pass through the program is used to set up a base for later testing of convergence. After the entry weight, entry module gross launch weight, total gross launch weight, entry module length and total configuration length have been defined, the program returns to the test on entry weight over planform area. On the next pass a new set of convergence parameters are defined. Convergence is achieved if the absolute difference between the two passes is less than an input decimal fraction of the last computed parameter. Entry weight is tested first, then entry module gross launch weight, and finally total configuration gross launch weight. When all of these have been met the size subroutine is called to define the length of the individual elements of the configuration. The size subroutine makes its computations based upon either volume requirements of the subsystems or minimum height constraints. Convergence is required of both the entry vehicle and the total configuration length. Final convergence is not achieved until all tests on weight and size are passed in sequence. The program can treat cargo weight as independent (as described above) or dependent. When cargo is dependent (the effective launch weight is fixed), the gross launch weight replaces the entry module gross launch weight as a convergence parameter, and the total launch cargo replaces the gross launch weight convergence parameter as discussed above.

After completion of all tests, the generate (GENERATE) subroutine is called to calculate a material distribution which is used in the costing program. Next the output (OUTPUT) subroutine is called to print the final convergenced results.

This completes the main logic flow of the program. Interrelationships for the logic are shown in Figure 6.1-2. The table shows inputs, the subroutine (engineering discipline equivalent), and the output. Note how the

FIGURE 6.1-2

WEIGHT/SIZING COMPUTER PROGRAM

INPUTS	SUBROUTINE	OUTPUT
Initial lengths - R.M. & M.M. Body Coordinates	Geometry	Wetted Areas Volumes Perimeters Hoop Tension Data
Lift/Drag Orbit Velocity Flight Path Angle Weight/Plan Area .	Aerodynamics (Entry Trajectory)	Velocity Altitude Density Time
Equation Selectors Aero Output Geometry Output	Temperature	Heating Rate Total Heat Heating Time Temperature
Concept Selectors Temperature Output Geometry Output	Thermal Protection System	Shingle Weight Insulation Weight Water Weight Ablation Weight
Crew Size Provisions per Man	Personnel	Crew Weight Provision Weight
Total Cargo Launched Distribution of Cargo (Effective Launch Weight)	Cargo	R.M. Launch and Return Cargo Weight M.M. Launch and Retro-graded Cargo Weight
Subsystem Coefficients Mass Properties Output	Landing	Subsystem Weight
Crew Size Heat Loads Volume	Environmental Control	Oxygen Loop Weights Coolant Loop Weight
Load Coefficients Geometry Output Mass Distribution Load Factors	Load	Moments, Axial Load and Shear at Launch and Landing
Load Output Geometry Output Material Properties	Structure	Rings and Shell Weight Structural Increments for Bulkheads, Hatches Floors, Windows

FIGURE 6.1-2
(Cont.)

WEIGHT/SIZING COMPUTER PROGRAM (Cont'd)

INPUTS	SUBROUTINE	OUTPUT
All Weight Output Geometry Output Subsystem Densities Subsystem Order	Mass Properties	Mission Weight History Ballast for Entry C.G. Mass Distribution Inertia, Products of Inertia, C.G. History
Performance ΔV and Impulses Propellant Properties Engine Configurations Mass Property Output Geometry Output	Propulsion	Propellant Quantities Engine Weights and Thrusts Tankage and System Weight
Geometric Shape Sizing Constraints	Size	R.M. Length M.M. Length
Power Requirements Specific Weights Geometry Output	Power	Power Generation System Weight Power Distribution System Weight Aero Control Power System Weight
Guidance and Control Component Wt. Communication Component Wt. Docking Definition Separation Prov. Coeff.	Miscellaneous	Subsystem Weight for G & N, Comm. On-Board Checkout-out, Range Safety, Docking, Separation, Recovery, Flotation
Structural Weights Thermal Prot. Weights Aero Surface Weights	Cost	Material Distribution for Use in Cost Analysis

data must flow between disciplines with the output from one routine being the input for the next.

6.2 PRSNL Subroutine - This subroutine calculates the weight of personnel and personnel provisions for the crew module and mission module. The weight requirements for each module are calculated separately. Input data is read into the DPSN array. Output is stored in the PSN array.

Personnel provisions for each module include:

1. Life Support System

   Food, first aid equipment, survival equipment (C.M. only) and waste system

2. Seats
3. Crew

   Men, suits, and personnel

   Parachutes (C.M. only)

4. Crew Accessories

6.2.1 Equation Discussion - The following equations compute the weights of the personnel and personnel provisions.

6.2.1.1 This equation defines the weight of men in the crew module. The selected weight per man is 200.8 pounds.

Equation 1: WMANC = XWTMAN * XNOMEN

WMANC = Weight of men in C.M. (lbs.)

XWTMAN = Average weight of men in C.M. (lb./man)

XNOMEN = Number of men launched in C.M. (men)

6.2.1.2 This equation defines the suit weight. If no information is available, 30 lb./man is used.

Equation 2: WSUITC = ASWMAN * XNOMEN

WSUITC = Weight of suits in C.M. (lbs.)

ASWMAN = Suit weight per man (lb./man)

XNOMEN = Number of men launched in C.M. (men)

6.2.1.3 This equation defines the personnel parachute weight. The first term, PPSEL = 0 if chutes are not required. PPSEL = 1 if chutes are required. Chute weight is based on 25 lb. per chute.

Equation 3: WCHUTE = PPSEL * 25. * XNOMEN

WCHUTE = Weight of suits in C.M. (lbs.)

PPSEL = Personal parachute selector

XNOMEN = Number of men launched in C.M. (men)

6.2.1.4 This equation defining food weight is based on 1.5 lb./man day of food and containers.

Equation 4: WFOODC = 1.5 * NOMEN * XTIME

WFOODC = Weight of food in C.M. (lbs.)

XNOMEN = Number of men launched in C.M. (men)

XTIME = Mission time sizing C.M.

6.2.1.5 This equation defines first aid equipment weight. The weight of equipment per man is a variable chosen by mission requirements and assumed to be 2.0 lbs./man in this model.

Equation 5: WFAIDC = FAIDC * XNOMEN

WFAIDC = First aid equipment weight in C.M. (lbs.)

FAIDC = First aid equipment weight/man in C.M. (lb./man)

XNOMEN = Number of men launched in C.M. (men)

6.2.1.6 The survival equipment weight defined by this equation depends on mission requirements and is assumed to be 2.5 lbs./man in this model. model.

Equation 6: WSURVC = SURVC * XNOMEN

WSURVC = Survival equipment weight in C.M. (lbs.)

SURVC = Weight/man of survival equipment in C.M. (lb./man)

XNOMEN = Number of men launched in C.M. (men)

6.2.1.7 The weight of the waste management system is based on 3 lb./man plus .2 lb./man day.

Equation 7: WWASTC = 3. * XNOMEN + .2 * XNOMEN * XTIME

WWASTC = Weight of waste system (lbs.)

XNOMEN = Number of men launched in C.M. (men)

XTIME = Mission time sizing C.M. systems (days)

6.2.1.8 The life support system weight is the sum of the food, first aid equipment, survival equipment, and waste management system weights.

Equation 8: WLSPTC = WFOODC + WFAIDC + WSURVC + WWASTC

WLSPTC = Life support system weight (lbs)

WFOODC = Equation 4

WFAIDC = Equation 5

WSURVC = Equation 6

WWASTC = Equation 7

6.2.1.9 This equation defines the crew accessories weight. It is based on 19 lb./man.

Equation 9: WCRASC = 19. * XNOMEN

WCRASC = Weight of crew accessories in C.M. (lbs.)

XNOMEN = Number of men launched in C.M. (men)

6.2.1.10 The seat weight is variable depending upon the type of seats used. Various seat weights are listed below:

Seat Type	Weight (Lb/Man)
Low "g"	22.0
Gemini Ejection Seat	152.0
Mercury Couch	72.0
Mercury Web Seats	35.0
Martin Baker (MK-5)	208.0
Stanley Encapsulated Seat	481.0
OCPDM Model	40.0

Equation 10: WSEATC = XSEAT * XNOMEN

WSEATC = Seat weight in C.M. (lbs.)

XSEAT = Weight of seat per man in C.M. (lb./man)

XNOMEN = Number of men launched in C.M. (men)

6.2.1.11 The crew weight is the sum of the suit weight, personal parachute weight (if required), and the weight of the men.

Equation 11: WCREWC = WMANC + WSUITC + WCHUTE

WCREWC = Weight of crew - C. M. (lbs.)

WMANC = Equation 1

WSUITC = Equation 2

WCHUTE = Equation 3

6.2.1.12 This equation defines the total weight of the crew systems. It is the sum of the crew weight, life support system weight, crew accessories weight, and seat weight.

Equation 12: WTOTC = WCREWC + WSLPTC + WCRASC + WSEATC

WTOTC = Total weight of C.M. (lbs.)

WCREWC = Equation 11

WSLPTC = Equation 8

WCRASC = Equation 9

WSEATC = Equation 10

6.2.1.13 This set of equations (13 through 22) apply to the mission module and are analogous to the first twelve equations which apply to the crew module. Two equations, the personal parachute and survival equipment equations, are left out of the mission module set of equations.

Equation 13: WMANM = YWTMAN * YNOMEN

Equation 14: WSUITM = ASWMAN * YNOMEN

Equation 15: WFOODM = 1.5 * ZNOMEN * YTIME

Equation 16: WFAIDM = FAIDM * ZNOMEN

Equation 17: WWASTM = 3.0 * ZNOMEN + .2 * ZNOMEN * YTIME

Equation 18: WLSPTM = WFOODM + WFAIDM + WWASTM

Equation 19: WCRASM = 19. * YNOMEN

Equation 20: WSEATM = YNOMEN * YSEAT

Equation 21: YCREWM = WMANM + WSUITM

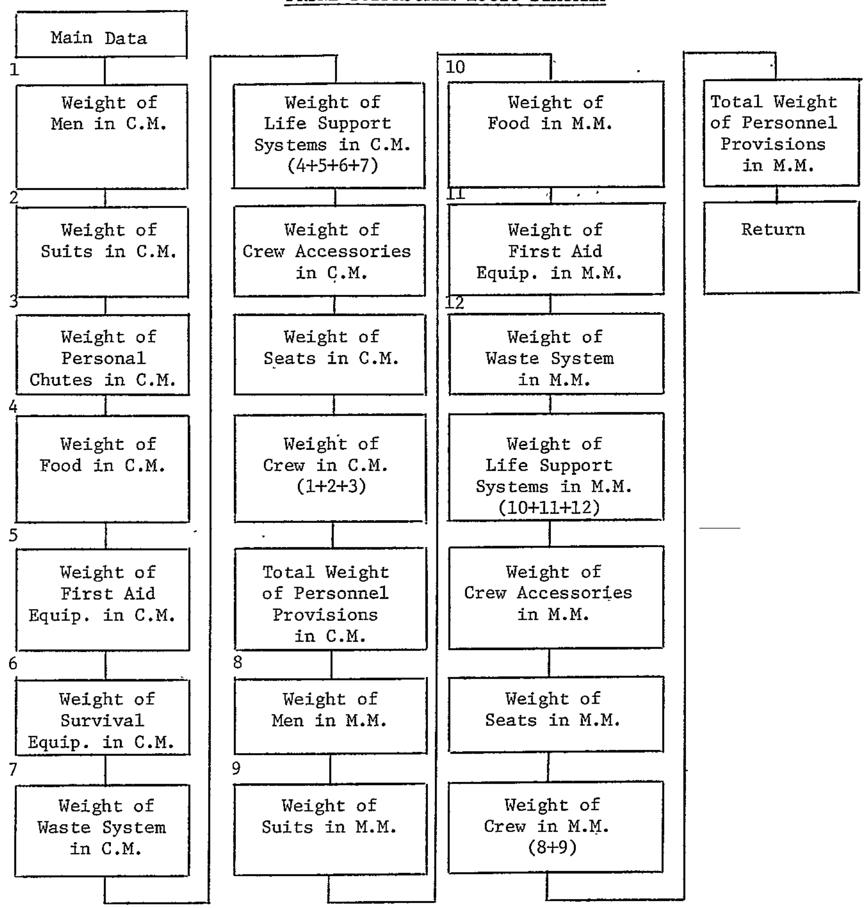
6.2.2 Flow Diagram - The logic flow of this subroutine is shown in Figure 6.2-1.

6.3 Geometry Subroutine - When the executive subroutine calls geometry (GEOM) the vehicle has already been sized to a basic overall length. The purpose of GEOM is to calculate various geometric data pertaining to the particular vehicle. These calculations are independent of the shape of the vehicle, with the restriction that it is symmetric about its vertical centerline.

The geometric analysis is done in four phases. Phases one and two calculate data pertaining to the entry module and its fins. Phase three sizes the mission module. Phase four wraps up the geometry package by performing

FIGURE 6.2-1

PRSNL SUBPROGRAM LOGIC DIAGRAM



various calculations that depend only on input data and on the overall length of the entry module.

The entry module is, for geometric purposes, divided into five sections: 1) nose, 2) forward, 3) center, 4) aft, and 5) the trailing section. For each section, the following information is calculated:

1) length,
2) inner moldline perimeter and cross sectional area of the art end of the section,
3) inner and outer surface areas of the bottom, side, and top,
4) enclosed and outer surface areas of the half-breadth radius,
5) enclosed and outer surface areas of the keel,
6) CG waterline value of the inner moldline at the aft end of the section, and
7) inner volume.

Also calculated are the total outer volume and total outer surface area.

The fin calculations are done in subroutines FING and FINML. For each of the 6 possible fins, the following items are calculated.

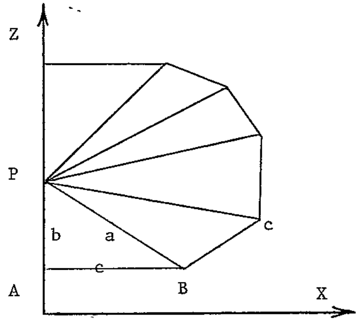
1) Outer volume
2) CG coordinates
3) Normal distance between CG and body
4) Root and tip chord lengths
5) Root cross sectional area and average thickness
6) Tru-view area of the fin
7) Tru-view area of the torque box
8) LE enclosed surface area, and
9) Coordinates of the intersection of the LE of the fin with the body.

If the fin is a fixed fin, the body station of the intersection of the trailing edge of the fin with the body is given. In the case of the movable fin, there is no such trailing edge intersection.

6.3.1 Geometry Equations - Each body station is described by the coordinates of at most 50 points. The equations used in GEOM give results that are as accurate as is the broken line approximation of the true moldline contour. Consider an arbitrary body station as illustrated in Figure 6.3-1.

FIGURE 6.3-1

TYPICAL BODY STATION



The perimeter of the section is taken to be the sum of the lengths of the straight line segments generating the section. The area of the section is calculated by summing the areas of the triangles which make up the section as indicated in the drawing. The formula used for the area of a triangle is $A = \sqrt{S(S-a)(S-b)(S-c)}$ where a, b, and c are the lengths of the sides of the triangle, and $S = (a+b+c)/2$.

In calculating the coordinates of the center of gravity (CG) of the section, the same component triangles are used. Each triangle is thought of as contributing a mass proportional to its area and located at its CG. If Z_i denotes the Z of the CG for the i^{th} triangle, and Ai represents the area of the i^{th} triangle, then

$$M_{\bar{Z}} = \frac{\sum_{i=1}^{n} Z_i A_i}{A}$$

where n is the number of component triangles and A is the total area of the section.

Volume calculations are made by Simpson's rule, that is,

$$PV = \frac{(\Delta Y)(A_1 + 4 * A_2 + A_3)}{6}$$

where ΔY represents half the distance between the body stations enclosing the partial volume (PV) being calculated, and A_1, A_2 and A_3 are the cross section areas at the forward, middle, and aft end of the section under consideration. The total volume is merely the sum of the partial volumes. As many as 54 cross section areas may be used in calculating the total volume.

For a given vehicle, the number of points used to define a body station remains constant. The surface area between two consecutive body stations is approximated by the panels found when points at one body station are connected to their counterparts at the next body station as indicated in the diagram Figure 6.3-2.

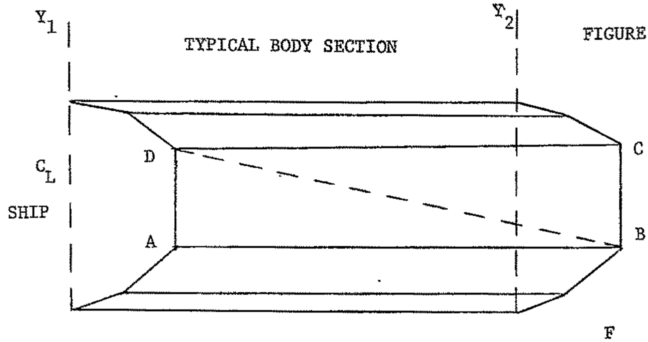


FIGURE 6.3-2

Consider the panel ABCD. Since all four of these points may not lie in the same plane, the panel is broken up into triangles DAB and BCD. The area of each triangle is calculated by the preceding area formula. The surface area of the panel is approximated by the sum of the areas of the two triangles, and the surface area of the section is considered to be the sum of the areas of its component panels.

Many of the calculations used in GEOM and SIZE are performed in one or more supporting subroutines. A list of these routines and an outline of their respective input/output parameters follows.

6.3.2 Subroutine APCA2 (N, NP, X, Y, Z, P, A, CX, CY, CZ) - The purpose of this routine is to calculate the perimeter, area, and centroid of a closed plane contour.

The input parameters are N, NP, X, Y, and Z. The routine works with points (XPT(I), YPT(I), ZPT(I)) where I ranges from N to NP. The coordinates of a point on or within the perimeter are given by (X, Y, Z).

The output elements P and A represent the perimeter and area of the closed curve. The point (CX, CY, CZ) is the centroid of the closed curve.

6.3.3 Subroutine Angle (A, C, S, T) - The purpose of this routine is to obtain the cosine, sine, and tangent of angle A where A is given in degrees. A is the input parameter. The output parameters are C (cosine), S (sine), and T (tangent) of the angle A.

6.3.4 Subroutine VEHML (KEY, OL, Y, ITEM, NPTS, T) - The input parameters of this routine are the values KEY, OL, Y, and ITEM. A value of 4, 5, or 6 for KEY indicates we are concerned with the Ballistic 60, Ballistic 40, or M2F2 vehicle. OL is the desired overall length of the vehicle, and ITEM indicates what type of information we want at the body station Y.

If ITEM = 1 (or 2) outer (or inner) moldline points will be calculated and NPTS is the number of points obtained at this Y-cut. T is then a dummy.

If ITEM = 3, the value of T becomes the floor to ceiling height at the given Y station. This height is measured at the centerline of the ship. NPTS is a dummy.

If ITEM = 4, the value of T becomes the outer moldline half breadth width at the given Y station. NPTS is a dummy.

For ITEM = 5, 6, 7, and 8, the following items are returned: The Z-value of the CG at the nose, the body leading edge radius at the Y-cut, the body keel radius at the Y-cut (not applicable for KEY = 4, 5, or 6), and the body nose sphere radius. These values are returned as T, and in each case, NPTS is a dummy.

None of these required calculations is actually done in VEHML. That is, VEHML merely calls the defining subroutine indicated by KEY, and the calculations are done in that subroutine. For example, if KEY is 6, VEHML calls Subroutine M2F2 (OL, Y, ITEM, NPTS, T) and Subroutine M2F2 does the actual calculations.

6.3.5 Subroutine FINML (KEY, N, OL, NPTS) - This subroutine does for the fins what VEHML does for the body. KEY indicates which vehicle we are working with, N is the number of the fin (N ranges from 1 thru 6), and OL is the present length of the vehicle. NPTS is the number of points at the root and tip sections used to define the fin contour. If the particular fin does not exist for the given vehicle, NPTS is assigned the value of zero.

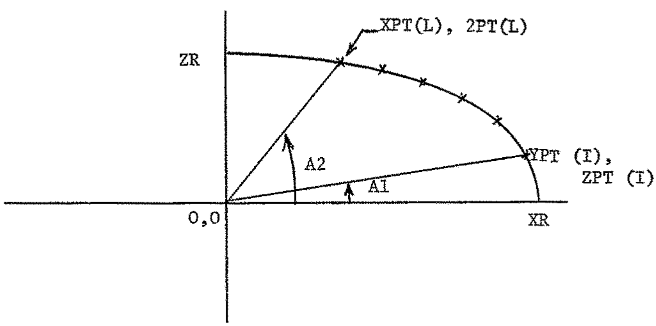
FINML does not calculate the point coordinates for each fin, rather, it calls the appropriate subroutine (as indicated by KEY and N), and the calculations are done there.

If NPTS is not zero, the fin exists and FINML will call the subroutine for fin geometry (FING). FING determines the geometric properties of the fin using the same volume, surface area, and CG formulae that GEOM uses.

6.3.6 Subroutine ELLSEG (Y, X, Z, XR, ZR, A1, I, XPT, YPT, ZPT, L, A2) - This routine calculates L - I + 1 points between angles A1 and A2 as indicated on the ellipse shown in Figure 6.3-3. In this example, the points are stored as (XPT, YPT, ZPT).

TYPICAL SEGMENT OF ELLIPSE

FIGURE 6.3-3

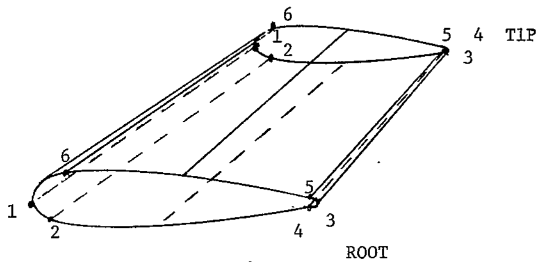


The subroutines BAL 40, BAL 60, and M2F2 are the actual defining routines for the various vehicles. Each routine is capable of generating all the information VEHML may be seeking. These routines were developed in such a way as to recreate the basic blueprint defining lines. The equations these routines use are dictated by the blueprint configuration.

6.3.7 Subroutine FING - Subroutine FING is designed to calculate and store the basic geometric parameters of a given fin. As indicated in the introduction, a total of sixty geometric points may be used to describe the contour of a particular fin. Thirty points define the root section, thirty points define the tip section, and straight line elements connect root points with their respective tip points. The general fin configuration is shown in Figure 6.3-4.

GENERAL FIN CONFIGURATION

FIGURE 6.3-4



Of the possible thirty points used to describe the root (or tip) contour, six of the points have special geometric significance. The leading edge of the fin is enclosed by points 1-2-6-1, the torque box is enclosed by points 2-3-5-6-2, and the trailing edge is enclosed by points 3-4-5-3.

The logic flow thru FING follows. The cross section area and the perimeter of the root is calculated. The span of the fin is calculated, where the span is measured from the center of the root section to the center of the tip section.

The volume of the fin is calculated based on cross section areas taken at the root, the tip, and midway between the root and the tip.

The true-view area and the CG of the fin are calculated based on the four sided figure formed by joining points 1 and 4 of the root with points 1 and 4 of the tip. The true-view area of the torque box is calculated based on the four sided figure formed by joining points 2 and 3 of the root with points 2 and 3 of the tip. The root (tip) chord length is measured from point 1 to point 4 of the root (tip).

The distance between the CG and the root of the fin is measured normal to a plane passing through the root chord and perpendicular to the plane formed by the leading edge and the root chord.

The enclosed leading edge area is the area enclosed by points 1-2-6-1 of the root and tip. The intersection of the leading edge of the fin, and the number of fins of this type are taken directly from input data.

6.3.8 Logic Flow - Before the basic flow of GEOM can be followed, it is necessary to describe the storage areas used by GEOM and its supporting routines.

The common area called VEHPTS stores the point coordinate information necessary to define a given body station, and is in the form: CØMMØN/VEHPTS/XPT(50),YPT(50),ZPT(50),NPT(8). If the most recent call to VEHML has requested outer (inner) moldline points at body station Y, for example, then (XPT(N),YPT(N),ZPT(N)) is the Nth point describing the outer (inner) moldline contour. YPT(N) will equal Y for all values of N. NTP(1) thru NTP(7) divide the section into its keel, bottom, half-breadth radius, side, and top components as indicated in the introduction. For example, if points 17 thru 24 are on the half-breadth radius, the NTP(3) would have a value of 17, and NTP(5) would have a value of 24. If 30 points are used to define the contour at each Y station, then NTP(7) will always equal 30.

The common area called VVEH has the form: CØMMØN/VVEH/PYS(30),NYS,NB(6),NBRK,NMC(30),NTMC and is used to store vehicle data that is not a function of any specific Y station. The PYS array stores the percent-Y-station values of the defining body stations. For example, if ØL represents the overall length of the vehicle, then PYS(N)*ØL is the numerical value of the Nth body station. NYS is the total number of Y stations used to define the vehicle. The NB array divides the vehicle into its 5 main geometric sections. That is, if the first five Y stations are used to define the nose of the vehicle, then NB(2)=5. NB(1) will always equal 1, and NB(6) will always equal NYS. NBRK is the total number of these break stations, and it will always be 6. The NMC array stores the subscript of the defining body stations at which a Mohr Circle is to be calculated. NTMC is the number of these stations. Thus, if the first Mohr Circle is to be calculated at the Nth defining Y station, NMC(1)=N. The numerical value of this Y station will be PYS(N)*ØL. Values are assigned to the VVEH elements when the first call to VEHML is made.

The GPDATA common area is used to store values pertinent to the Mohr Circle (hoop) calculations. The first element in this area indicates which hoop station is being processed. The second element is the number of points used to approximate the inner moldline at the particular hoop station. The third element is the number of the defining Y station of the hoop station. Suppose the common area GPDATA has the form COMMON/GPDATA/N,NPTS,NY,BSTA. Suppose also that OL represents the overall length of the vehicle and we are at body station PYS(NY)*ØL. If, for example, NY=NMC(3) a call is made to subroutine HOOP to calculate Mohr Circle data for the third hoop station (which is also the NYth defining Y station). BSTA is set equal to PYS(NY)*ØL.

The VGEØM common is used only for temporary storage. The first element of this common should always be zero for the M2F2 and ballistic cases.

The conventions followed by GEOM will be mentioned as they are encountered in following the flow of GEOM. For example, in GEOM as well as all of its supporting routines, OL is used to designate the overall length of the vehicle being analyzed. Also, KEY is assigned the input value used to indicate which vehicle is being considered.

The first call to VEHML is made to obtain the body leading edge radius, the body nose sphere radius and the CG of the nose. The first call to VEHML will also initialize the values in the VVEH common.

A series of calls are made to VEHML to establish inner moldline points at each of the body break stations. There are six (NBRK) of these stations, but since the first station is at the nose tip and thus is only a point, this station is omitted. After VEHML has returned the inner moldline points at a particular break station, APCA2 is called to operate on these points. As mentioned earlier, APCA2 will calculate the perimeter, cross-sectional area, and centroid of this body station.

The values returned by VEHML will have inches as their units. Moreover, items such as inner moldline point coordinates, outer moldline point coordinates, and half-breadth widths are calculated for one side (left hand looking aft) of the vehicle. For example, the area computed by APCA2 has units of square inches and is actually the area for only one half of the vehicle. The unit of measure for the output array of GEOM (the G-array) is feet, so the area returned by APCA2 must be doubled and then divided by 144 before it can be stored in the G array. Adjustments of this kind are made throughout the subroutine whenever the unit being operated on is not compatible with the output unit.

The next loop of calls to VEHML is used to calculate data on the inner moldline of the ship. If the body station used in calling VEHML is a Mohr Circle station, subroutine HOOP is called. In subroutine HOOP, Mohr Circle data is calculated based on the inner moldline points just returned from the last call to VEHML. Volume and surface area are calculated next. In the interest of accuracy, not only the NYS defining body stations, but also one body station midway between each set of two consecutive defining stations is used in these calculations.

The next series of calculations pertains to the outer moldline of the vehicle. In this case outer moldline calculations use body stations other than the NYS defining stations in the same way that they were used for inner moldline calculations. Throughout this series of calls to VEHML, checks are made to find the maximum cross sectional area. Planform area is computed using the point [XPT(NTP(8)), YPT(NTP(8)), ZPT(NTP(8))] to describe the planform perimeter. Surface area is calculated by sections; that is, keel, bottom, half-breadth radius, side, and top surface areas are all calculated separately. Volume is calculated based on the total cross section. The calculated data is stored in the output array (G-array).

When the outer moldline calculations are complete, the fins are processed by making a series of six calls to subroutine FINML. If fin number N (N = 1, 2, 3, 4, 5, 6) exists, the FINML calls subroutine FING. FING calculates and stores all the required fin geometry. This completes the geometric analysis of the entry module and the sizing of the mission module begins.

If the input data does not specify a maximum diameter for the mission module cone, then there will be no mission module cylinder and there will be no trailing cone. Instead, the mission module cone is sized to the length the required volume dictates.

If a maximum diameter for the mission module is specified, and the required diameter at the booster is not specified, then the mission module cone length is fixed, the mission module cylinder length is set to zero, and the trailing cone is sized to meet the remaining volume requirements.

If both the maximum diameter for the mission module and the diameter at the booster are specified, and the mission module is not to be sized for volume, then the cylinder length is determined by input data. If, however, the mission module is to be sized for volume, then the length of the cylinder is adjusted to meet the volume requirement.

When the sizing of the mission module is complete, its surface area and volume are calculated. The remaining portion of the program loads the output array with values that are direct functions of entry module length or mission module length.

The logic flow is shown in Figure 6.3-5.

6.4 Aero Subroutines

6.4.1 General Approach - The aerodynamics subroutine provides entry trajectory time histories from which heat protection requirements can be derived. The trajectories are calculated using closed form solutions of the minor circle turn entry profile. The equations are appropriate primarily for lifting type spacecraft executing a maximum crossrange entry. A second mode provides a slightly different bank angle schedule that minimizes the downrange while maintaining a high crossrange.

A wide range of heating environments can be attained, depending on the selected entry conditions. Considerable discretion must be used to obtain a trajectory profile suitable to the specified missions and spacecraft type. The currently programmed aero routine provides only two trajectories, one in the maximum crossrange mode (Mode I) and one in the minimum downrange mode (Mode II), see reference 6.4-1. Input parameters can be selected to provide the most appropriate conditions on which to base the design of the heat protection system. Entry flight path angle, angle of attack, lift coefficient, lift to drag ratio, and minor circle turn radius can be arbitrarily selected for each mode.

For lifting body heat protection design, the maximum total heat is generally attained with a shallow entry at maximum L/D and zero bank angle.

FIGURE 6.3-5

GEOM

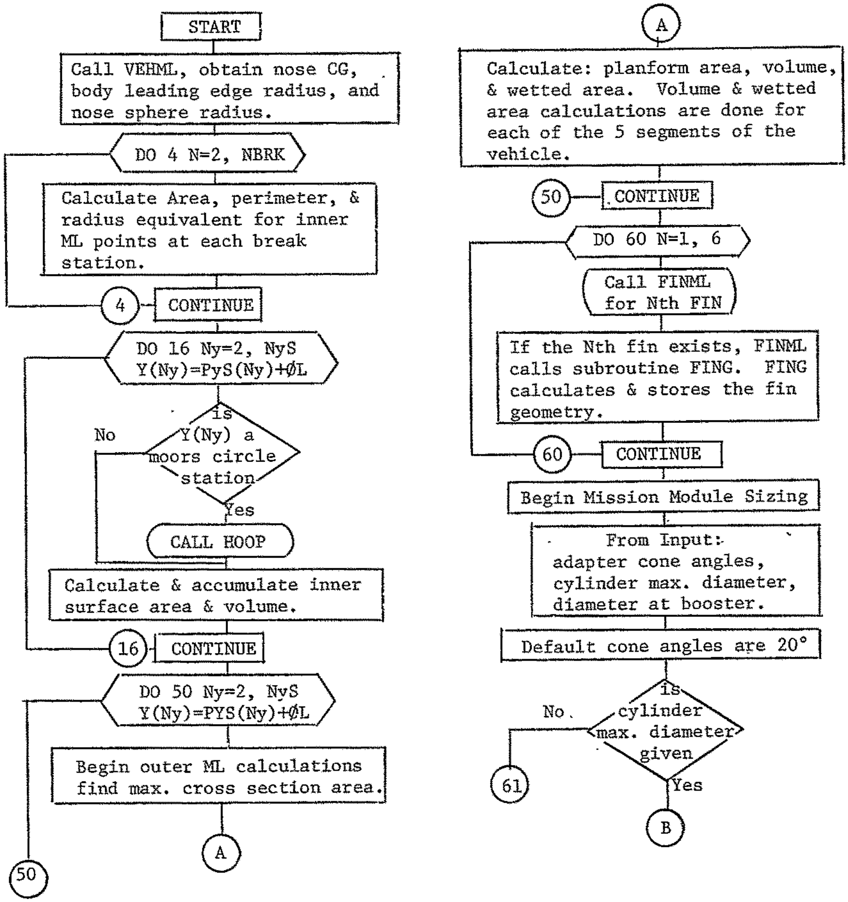
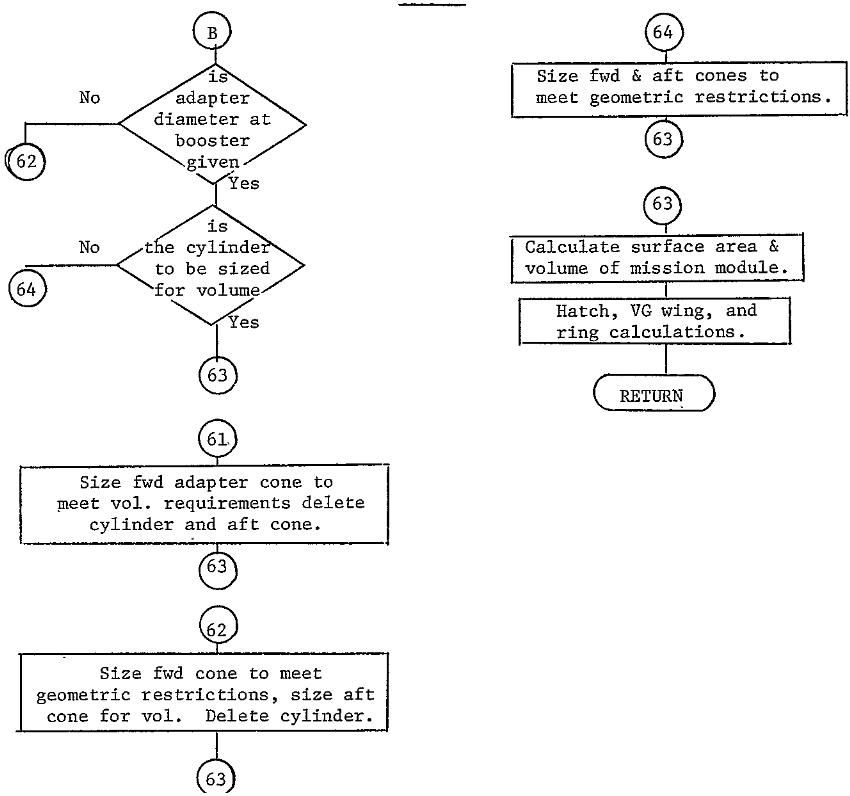


FIGURE 6.3-5
(Cont.)

GEOM



Maximum heating rates on the other hand, are attained with steep entry angles and high bank angle but not necessarily at either minimum or maximum angle of attack. Since the sizing model aero routine uses a programmed variable bank angle schedule, the trajectories cannot directly simulate either the maximum total heat or maximum heating rate conditions. Therefore, input parameters must be selected that will approximate as closely as possible the desired trajectories for heat protection design.

For a lifting body configuration the maximum total heat trajectory is closely approximated by using a Mode I entry at the minimum expected entry flight path angle and an angle of attack for maximum L/D. Maximum heating rates are approximated by using a Mode II entry trajectory at the steepest expected flight path angle and angle of attack for maximum lift. The flight path angle selected must be adjusted as discussed in Section 6.4.3 to attain the correct pull out altitude.

For a lifting ballistic spacecraft configuration, the maximum total heat is again approximated with a Mode I entry at maximum L/D and minimum entry flight path angle. For maximum heating rate, a zero L/D entry is necessary. However, since the programmed minor circle trajectory is not adaptable to a zero lift condition, this trajectory must be approximated by using the minimum feasible lift with the steepest expected entry flight path angle.

6.4.2 Equation Derivations - The entry trajectories are calculated using a closed form solution of the minor circle entry profile. This analytic solution is unique, and relatively simple, because the spherical equations of motion were avoided in favor of a circular equation involving the minor circle turn angle, σ, which is distinct from the heading angle, ψ. The results agree with a computer machine solution programmed to follow the specified bank angle vs altitude history.

The derivation of the equations of motion was performed with non-dimensional parameters so that the results are applicable to any planet or atmosphere. The equations may be used for all values of turn radii, as given by the planet central angle, λ.

Euler's equation from the calculus of variations was used to show that the gliding reentry along a minor circle gives the maximum cross range provided a specific bank angle schedule vs altitude is followed. Furthermore, a slightly different schedule will give minimum down range which also occurs in a minor circle turn.

6.4.2.1 Summary of Equations - The derivation of the closed form of the entry trajectory naturally divides into two parts. The first part is the wings level pull out from entry flight path angle, γ_o, to level flight, $\gamma = 0$. The second part is the banked turning glide from pull out to landing.

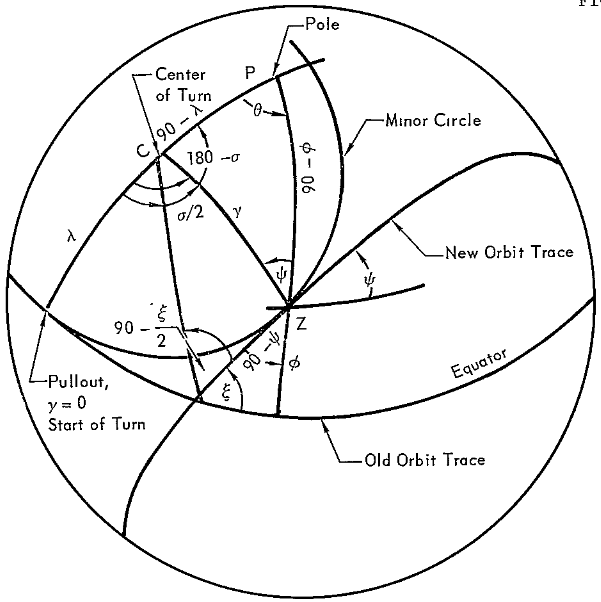
NOMENCLATURE (See Figure 6.4-1)

Subscripts

$(\ )_o$	Pre pull out orbital condition
$(\ )_p$	Pull out, $\gamma = 0$, initial turn condition
$(\ )_{90}$	At $\psi = 90°$, north heading
A	Aerodynamic reference area, ft^2
C_D	Drag coefficient
C_L	Lift coefficient
D	Drag, lbs
g	Acceleration of gravity, 32.2 fps^2 at R
L	Lift, lbs
m	Mass of spacecraft, $slugs/ft^3$
r	Minor circle turn radius, nm
R	Earth's radius, 3440 nm
t	Time, secs
V	Velocity, fps
W	Weight of spacecraft, lbs
β	Bank angle, measured from horizontal, degs
γ	Flight path dive angle, positive angle measured down from horizon, degs
η	$(V/Vo)^2$
θ	Longitude, measured from start of turn at pull out, degs
λ	Earth central angle of minor circle radius, radians
Λ	Atmospheric scale height (1/27,200 ft)
ρ	Air density, $slugs/ft^3$
σ	Minor circle turn angle, degrees
ϕ	Latitude, degs
ψ	Azimuth, heading measured from latitude, line, degs

MINOR CIRCLE TURN GEOMETRY

FIGURE 6.4-1



From CZP

$$\sin\phi = \sin\lambda \cos\lambda\,(1-\cos\sigma)$$

$$\tan\psi = \frac{\cos\lambda \sin\sigma}{1-(1-\cos\sigma)\cos^2\lambda}$$

$$\tan\theta = \frac{\sin\lambda \sin\sigma}{1-(1-\cos\sigma)\sin^2\lambda}$$

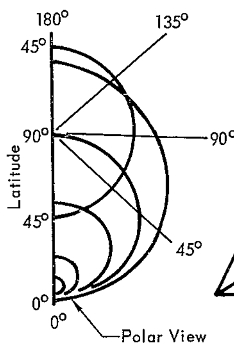
$$\frac{\sin\psi}{\cos\lambda} = \frac{\sin\sigma}{\cos\phi} = \frac{\sin\theta}{\sin\lambda}$$

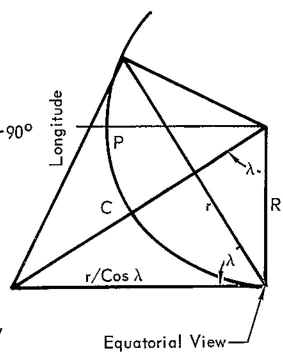
ϕ = Latitude
θ = Longitude (from Start of Turn)
ψ = Azimuth (from Equator)
σ = Minor Circle Turn Angle
λ = Earth Central Angle of Minor Circle Radius
ξ = Plane Change

$\sin\lambda = r/R$
r = Minor Circle Turn Radius
R = Earth Radius

$$\sin\phi = \frac{\tan\lambda - \cos\theta\sqrt{\tan^2\lambda - \sin^2\theta}}{\tan^2\lambda + \cos^2\theta}$$

Equatorial – Tangent Minor Circle





Pull Out Derivation - The γ vs h (or ρ) history is obtained with two simplifying assumptions. First, it is assumed that the entry flight path angle is small, and second, consistent with the first assumption, that the speed loss during the pull out is small compared to orbital speed. It is also assumed that entry begins at orbital speed, and that constant angle of attack (constant C_L) is held through the pull out. Then the lift equation of turning motion is

$$MV_o \frac{d\gamma}{dt} = -C_{L_o} A \frac{\rho}{2} V_o^2 \qquad 6.4.2\text{-}1$$

The independent variable, time, is changed to altitude, h, with the definition of the flight path angle.

$$\frac{dh}{dt} = -\gamma V_o \qquad 6.4.2\text{-}2$$

Substituting 6.4.2-2 into 6.4.2-1 gives

$$\gamma d\gamma = \frac{\rho_p}{2} \frac{C_{L_o} A}{m} \frac{\rho}{\rho_p} dh \qquad 6.4.2\text{-}3$$

After substituting the exponential atmosphere relation for ρ/ρ_p

$$\gamma d\gamma = \frac{\rho_p}{2} \frac{C_{L_o} A}{m} e^{-\Lambda(h-h_p)} \qquad 6.4.2\text{-}4$$

After integrating, and demanding that $\gamma = 0$ when $\rho = \rho_p$

$$\gamma^2 = \frac{C_{L_o} A \rho_p}{\Lambda m} \left(1 - \frac{\rho}{\rho_p}\right) \qquad 6.4.2\text{-}5$$

Since $\rho \cong 0$ at entry where $\gamma = \gamma_o$, Equation 6.4.2-5 is written more simply as

$$\left(\frac{\gamma}{\gamma_o}\right)^2 = 1 - \frac{\rho}{\rho_p} = 1 - e^{-\Lambda(h-h_p)} \qquad 6.4.2\text{-}6$$

The time history of the altitude (ρ) is obtained by substituting Equation 6.4.2-6 into 6.4.2-2

$$\gamma_o V_o \, dt = \frac{-dh}{1-e^{-\Lambda(h-h_p)}} \qquad 6.4.2\text{-}7$$

After integrating, and requiring that time be measured from pull out, $t = 0$ at $h = h_p$, which implies that time during pull out is negative, the result is

$$\Lambda V_o \gamma_o t = \ln \left(\frac{1 - \sqrt{1 - \frac{\rho}{\rho_p}}}{1 + \sqrt{1 + \frac{\rho}{\rho_p}}} \right) \qquad 6.4.2\text{-}8$$

Derivation of Minor Circle Trajectory - The minor circle turn trajectory is also derived by assuming a constant angle of attack, constant C_L and C_D. For equilibrium of forces in the vertical direction

$$\frac{L}{W} \cos \beta = 1-\eta \qquad 6.4.2\text{-}9$$

and, from Figure 6.4-1, equilibrium of the horizontal forces in a turn, with a constant turning radius of $r/\cos \lambda$, requires that

$$\frac{L}{W} \sin \beta = \frac{\eta}{\tan \lambda} \qquad 6.4.2\text{-}10$$

The bank angle, β, is eliminated from Equations 6.4.2-10 and 6.4.2-11 to give the load factor normal to the flight path.

$$\frac{L}{W} = \frac{C_L A}{} \ \frac{\rho}{2} \ V^2 = \left[\frac{\eta^2}{\tan^2 \lambda} + (1-\eta)^2 \right]^{1/2} \qquad 6.4.2\text{-}11$$

Since for a lifting body the normal force is usually much greater than the axial force, a good approximation of the normal load factor is

$$\frac{N}{W} \cong \sqrt{\left(\frac{L}{W}\right)^2 + \left(\frac{D}{W}\right)^2} = \frac{L}{W} \sqrt{1 + \frac{1}{(L/D)^2}} \qquad 6.4.2\text{-}12$$

Substituting 6.4.2-12 in 6.4.2-11 gives

$$\frac{N}{W} = \frac{\eta}{\tan \lambda} \sqrt{\left[1 + \frac{1}{(L/D)^2} \right] \times \left[1 + \left(\frac{1-\eta}{\eta} \tan \lambda\right)^2\right]} \qquad 6.4.2\text{-}13$$

The maximum load factor for a constant radius turn occurs at the maximum speed, $\eta = 1$, so that

$$\frac{N}{W}_{max} = \frac{1 + \frac{1}{(L/D)^2}}{\tan \lambda} \qquad 6.4.2\text{-}14$$

The minor circle turn radius parameter, tan λ, a constant, is found from Equation 6.4.2-11 at pull out where $\eta = 1$ and $\rho = \rho_p$.

$$\tan \lambda = \frac{2(m/C_L A)}{\rho_p R} \qquad 6.4.2\text{-}15$$

Tan λ is related to the entry angle, γ_o, by Equation 6.4.2-5 with $\rho = 0$.

$$\tan \lambda = \frac{2}{\Lambda R_{\gamma_o}{}^2} \quad \frac{C_{L_o}}{C_L} \qquad 6.4.2\text{-}16$$

The speed vs altitude relation is obtained by combining Equations 6.4.2-11 and 6.4.2-15 to give

$$\frac{\rho}{\rho_p} = \sqrt{1 + (\frac{1-\eta}{\eta} \quad \tan \quad \lambda)^2} \qquad 6.4.2\text{-}17$$

Time is related to these parameters through the equation of motion along the flight path.

$$m \frac{dV}{dt} = -C_D A \frac{\rho}{2} \quad V^2 \qquad 6.4.2\text{-}18$$

When this equation is solved for ρ and substituted into 6.4.2-11, the result is

$$\frac{g}{Vo} \frac{dt}{d\eta} = \frac{-\frac{1}{2}\frac{L}{D}}{\eta_\gamma \sqrt{(\frac{\eta}{\tan \lambda})^2 + (1-\eta)^2}} \qquad 6.4.2\text{-}19$$

with $t = 0$ at $\eta = 1$

6.4.2.2 Substantiating Data

Assumptions - The trajectory solution exploits the simplification obtained by assuming

(1) non rotating earth and atmosphere
(2) $\gamma \ll 1$, small flight path angles
(3) L/D constant with altitude and speed
(4) pullout from entry to $\gamma = 0$ with velocity losses very small compared to orbital speed.

Assumption (1) will yield somewhat conservative results for heading in the easterly directions, and correspondingly optimistic results in the westerly directions.

Assumption (2) is also conservative since no advantage is obtained from the potential energy available at pull-out altitudes.

Assumption (3) is not too severe since the pull-out altitudes are usually near 200,000 ft where the changing viscous effects are small. Furthermore the greater fraction of the maneuver is performed at supersonic and hypersonic velocities where there is a weak Mach variation.

Assumption (4) only directly affects the initial condition for the turn. It is assumed that the spacecraft is retrograded out of orbit to provide an entry angle, γ_o, at orbital speed. The pullout to level flight, from where the turn is begun, will cause some loss in speed which is neglected.

It is believed that any loss of L/D, and/or speed in the pull out, will be largely balanced by the neglected potential energy of Assumption (2).

A complete derivation of the ground track relation and other performance items is given in Reference 1.

6.4.3 Application of Engineering Data Input - Trajectories obtained using the aero routine require relatively few input parameters. However, since a wide range of heating environments can be attained, care must be used to select input parameters that give a trajectory profile suitable to the spacecraft mission.

As currently programmed, two trajectories are computed for each case, using variations of the minor circle entry profile. In the first (Mode I) a bank angle program providing maximum crossrange is utilized. The second trajectory (Mode II) provides a slightly different bank schedule to decrease the downrange while maintaining a high crossrange.

The Mode II entry profile can be used to simulate trajectories having high heating rates by using the maximum expected entry flight path angle. The most critical angle of attack for maximum heating rate is not apparent, however. For lifting body type spacecraft, the lower surface heating rate at a given altitude increases with angle of attack. However, the resultant increase in glide altitude compensates for this effect and decreases the heating rate. Upper surfaces, on the other hand, will generally exhibit maximum heating rate at minimum angle of attack, due to the lower glide altitude.

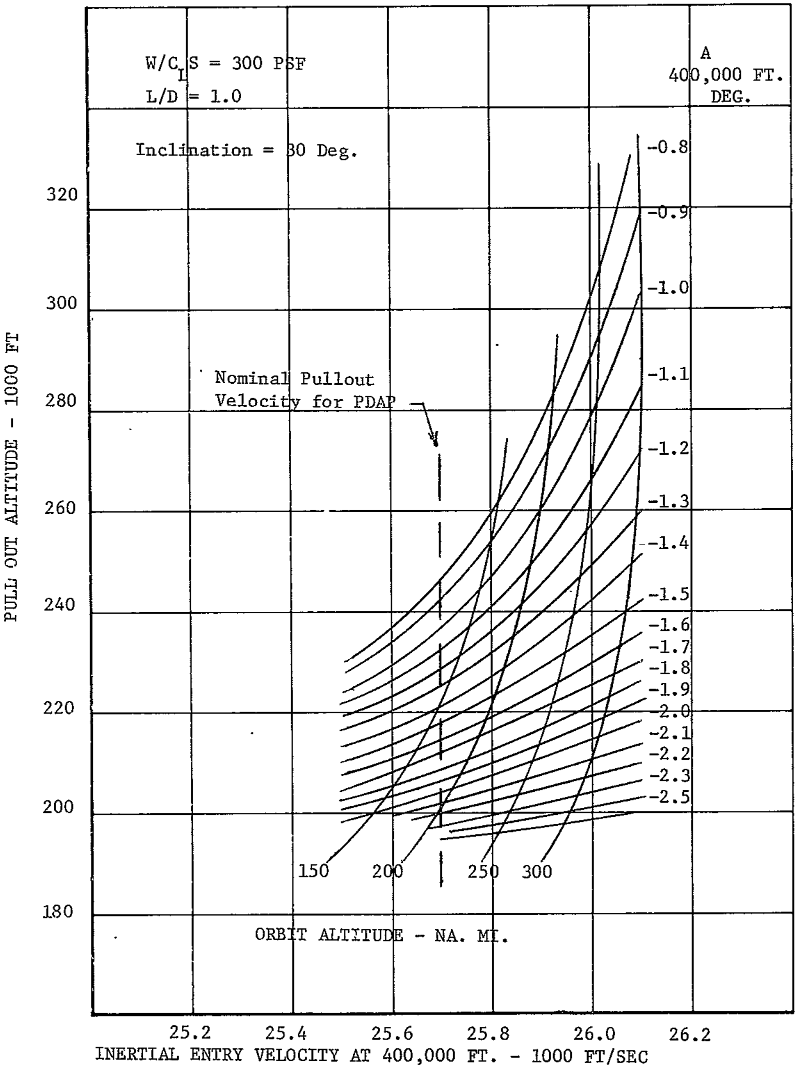
The minor circle entry trajectory is not inherently suited to ballistic type entry spacecraft since this type generally does not modulate bank angles for altitude control. Also, due to their relatively low lift-to-drag ratio, the assumption of a negligible velocity loss during pullout is no longer valid. In addition, for a ballistic spacecraft executing a zero lift entry, no clear cut pullout altitude is defined and a bank modulation program is meaningless. For these reasons, ballistic vehicle trajectories can only be approximated.

The maximum total heat trajectory, again, is best simulated with a Mode I entry using the minimum flight path angle and maximum L/D. Zero lift trajectories, which provide the maximum heating rates on a ballistic spacecraft, cannot be simulated with this program. For these cases, the trajectory can be only grossly approximated by using the steepest expected entry angle and the minimum L/D at which the program will operate, (approximately 0.2).

In selecting the initial flight path angle for both the ballistic and lifting spacecraft, some adjustments are necessary to obtain the proper trajectory. The pullout equations assume that the velocity during pullout is maintained constant at a nominal orbital velocity of 25,700 ft/sec. However, for entries from the higher orbital altitudes, the entry velocity can be somewhat greater than 25,700 ft/sec, thus providing a higher pullout altitude and significantly affecting the heating profile. To compensate for this, an equivalent entry angle can be selected that will provide the same pullout altitude using the nominal entry velocity as would be obtained with the actual entry conditions. Figure 6.4-2 illustrates a typical variation of pullout altitude with entry velocity for a range of entry flight path angles. The corresponding initial orbit altitude is also indicated. Although this data is for an entry vehicle having a $W/C_L S$ of 300 lb/ft^2, an L/D of 1.0 and 30° orbit inclination, the trends are similar for other values of these parameters. To determine the proper entry angle to use in the program, select the desired value of orbit altitude and entry angle from this plot, then move horizontally across on a constant altitude line until intersecting the nominal pullout velocity of 25,700 ft/sec. The flight path angle, γ at this point will then give the desired pullout conditions. For example, a spacecraft entering from a 250 NM orbit with an entry angle of -1.5 degrees will reach a pullout altitude of

Figure 6.4-2

EFFECT OF ENTRY VELOCITY ON PULLOUT ALTITUDE



232,000 ft at a velocity of 25,950 ft/sec. By using an entry angle of about -1.1 degrees with the nominal programmed velocity of 25,700 ft/sec the proper pullout altitude will be attained.

Typical trajectories obtained using this correction are shown in Figure 6.4-3 compared to trajectories computed using a point mass option of the McDonnell Douglas six degree of freedom generalized computer program. (Reference 6.4-2.) Results are shown for both the M2-F2 and ballistic configurations. For the lifting body, the trajectory simulation was programmed to maintain a constant altitude after pullout by modulating the bank angle from near 90 degrees until the desired bank was attained. This bank angle was then maintained for the remainder of the trajectory. Results for both the shallow entry at maximum L/D and the steep entry at maximum lift coefficient are seen to be in good agreement with the minor circle trajectories computed by the program.

For the ballistic configuration, the SDF trajectories were computed using a constant bank angle without altitude control. Although the minor circle trajectories are less well suited to a low L/D ballistic vehicle, fairly good agreement is shown with the SDF computed flight path.

Other input parameters needed are the minor circle turn radius (λ), reference altitude, and density at the reference altitude. The area loading, W/S, is derived during each iteration of the sizing model program.

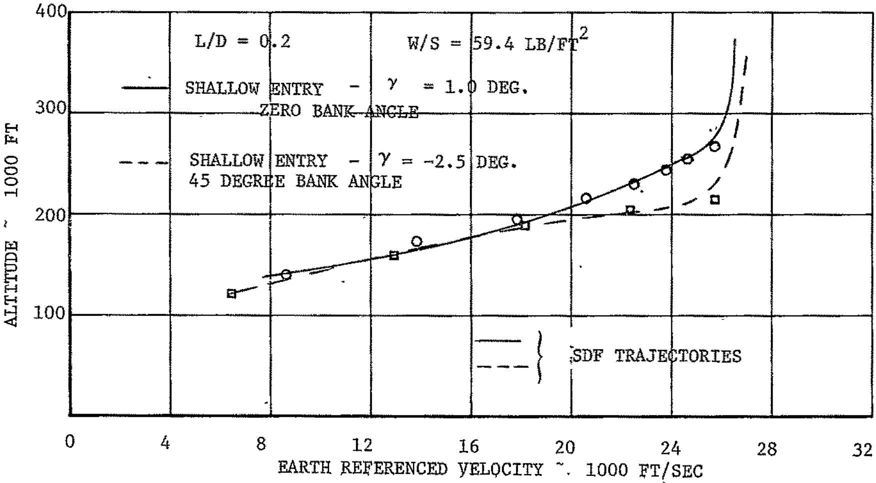
The minor circle turn radius, λ, is related to the initial entry angle, λ_o, as shown in equation 6.4.2-16. These two parameters therefore cannot be independently selected although some latitude is available by using a different lift coefficient during the pullout maneuver than during the remainder of the entry glide. This is accomplished by varying the ratio of C_{L_o} to C_L in equation 6.4.2-16, within the limits of the trim capability of the spacecraft.

The reference altitude and density input is needed because the program uses an exponential atmosphere. Altitude density relationships at other altitudes will therefore differ somewhat from the standard atmosphere. For this reason it is recommended that the reference altitude be selected near the critical heating altitude of the spacecraft.

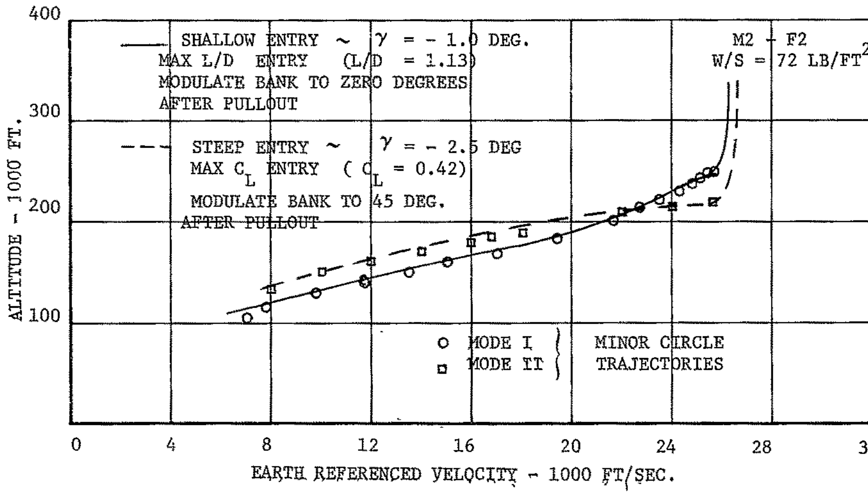
FIGURE 6.4-3

ENTRY TRAJECTORY

SEMI - BALLISTIC



ENTRY TRAJECTORIES



6.5 Temperature Subroutine - The TEMP subroutine uses approximate methods to calculate the worst combination of maximum temperatures, total heat, and heating duration at 27 body stations for a flight envelope defined by two trajectories. The steps in the heat transfer analyses at each of the 27 locations are as follows:

(1) At any point in the trajectory, the subroutine determines whether the boundary layer flow is laminar or turbulent.

(2) If laminar - the following empirical correlations are available:
- o Stagnation point (or constant ratio method)
- o Flat plate or cone
- o Ratio of local to stagnation point (with angle of attack variation)
- o Swept cylinder
- o Separated flow

(3) If turbulent - the following methods can be used:
- o Sonic line (or constant ratio method)
- o Flat plate or cone
- o Ratio of local to sonic line (with angle of attack variation)

The remaining steps in the thermal environment estimation are:

(4) Apply hot wall correction

(5) Convert heat fluxes to equilibrium temperatures

(6) Determine maximum surface temperature

(7) Integrate hot wall heating rate history to find total heat

(8) Compute heating duration

The relationships, logic and rationale for employing these empirically derived methods for rapid calculations are described herein. Where possible, the development of the approximate techniques, the assumptions introduced in simplifying the refined methods and the degree of correlation obtained with results from the more refined techniques are presented for each pertinent equation.

NOMENCLATURE

a	Speed of sound (ft./sec.)
ASSET	Aerothermodynamic/Elastic Structural Systems Environmental Tests
C_p	Specific heat (Btu/lb_m °R)
$[F(\delta_{eff})]_L$	Defined by Equation (4-3)
$[F(\delta_{eff})]_T$	Defined by Equation (5-3)
FDL-7MC	Lifting reentry spacecraft studied in MHTV contract
g	Standard acceleration of gravity (32.17 ft./sec.^{2})
H	Enthalpy (Btu/lb_m)
J	Mechanical equivalent of heat (778 ft-lb_f/BTU)
j	Input multiplier for heating equations
m	Mach number
MHTV	Manned Hypersonic Test Vehicle
MRS	Multipurpose Reusable Spacecraft
OC/PDM	Optimized Cost/Performance Design Methodology
P	Pressure (lb/ft^2)
PDAP	Preliminary Design Analyses Program
$\dot{q}$	Heating rate (Btu/ft^2 sec)
Q	Total heat (Btu/ft^2)
R	Radius (ft.)
R_e	Reynolds number
T	Temperature (°F or °R)
TRANS (I)	Boundary layer flow option, Figure 2
V	Velocity (ft./sec.)
X	Characteristic or wetted distance (ft.)
α	Angle of attack (degrees)
β	Compliment of geometric sweep angle measure in the pitch-roll plane (degrees)
$\propto$	Varies as
γ	Specific heat ratio
ε	Emittance
λ	Geometric Sweep angle (degrees)
δ	Flow deflection angle between surface and reference planes

μ	Viscosity ($lb._m$/ft. sec.)
ρ	Density (lb_m/ft^3 or $slugs/ft^3$)
σ	Stefan - Boltzmann constant (.4758 x 10^{-12} BTU/ft^2 sec °R^4)
τ	Time (seconds)
ϕ	Angle between the pitch-roll plane and the plane tangent to the surface (degrees)
ψ	Defined by Equation (3-10)

SUBSCRIPTS

∞	Freestream
θ	Based on momentum thickness
CW	Cold wall
EFF	Effective
L	Local or laminar
r	Reference
s	Stagnation
T	Turbulent
W	Wall

SUPERSCRIPTS

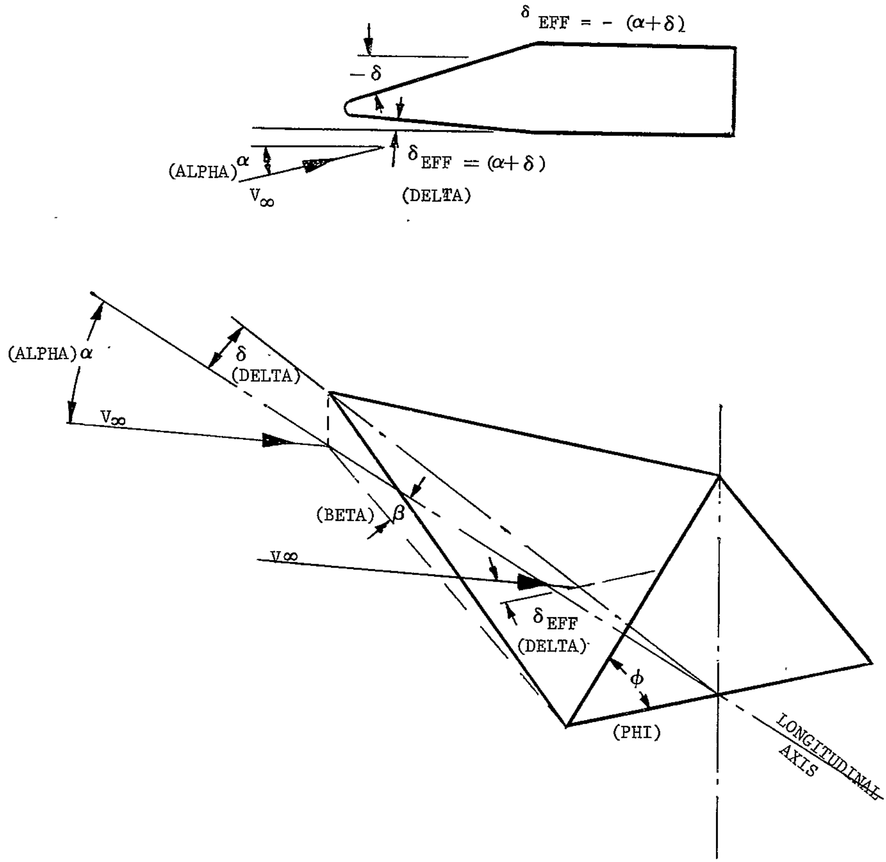
*	Based on reference enthalpy conditions (Eckert's method)

6.5.1 Vehicle Geometry Consideration - The vehicle configuration must be defined prior to performing the heat transfer analysis in the TEMP subroutine. For each of the 27 locations comprising the spacecraft the geometry properties needed are: (1) the flow deflection angles between the surface and reference planes, and (2) reference body lengths, nose cap and leading edge radii. The analysis is performed at the midpoint of each section with reference length values provided by the geometry subroutine. The fin side reference lengths are inputs.

The angles used to define the vehicle geometry are defined in Figure 6.5-1. The following relationship is utilized to calculate the effective flow deflection angle between the velocity vector and surface at angle of attack.

FIGURE 6.5-1

VEHICLE GEOMETRY DEFINITION



$$\delta_{EFF} = \sin^{-1}\left\{\frac{\cos\alpha\ (\sin\phi\ \sin\beta\ \cos\delta - \cos\phi\ \cos\beta\ \sin\delta) - \sin\alpha\ \cos\phi\ \cos\beta\ \cos\delta}{[\cos^2\beta\ \cos^2\delta + (\sin\phi\ \sin\beta\ \cos\delta - \cos\phi\ \cos\beta\ \sin\delta)^2]\ 1/2}\right\} \qquad 6.5.1\text{-}1$$

The effective sweep angle for a leading edge is determined from:

$$\lambda EFF = \sin^{-1}[\sin\lambda\ \cos\alpha + \cos\lambda\ \sin\phi\ \sin\alpha] \qquad 6.5.1\text{-}2$$

If erroneous input yields a negative λ_{EFF}, the program sets λ_{EFF} equal to 0. These effective angles are utilized in the transition and heat flux calculations, discussed in the following sections.

6.5.2 Boundary Layer Flow Transition - The largest uncertainty in heat transfer analysis for earth orbit entries is determining when, and how abruptly, transition from laminar to turbulent boundary layer flow occurs. Because detailed treatment of boundary layer transition is beyond the scope of this study, the approach followed was to start with the transition criteria perviously used in MDAC advanced design studies and to simplify them in order to arrive at relationships that are amenable to rapid calculations. The following transition criteria, based on correlation of cone and flat plate wind tunnel and flight data, have been utilized in previous studies.

$$\frac{R_{e_\theta}}{M_L} = 150 \qquad \text{flat plate} \qquad 6.5.2\text{-}1$$

$$\frac{R_{e_\theta}}{M_L} = 200 \qquad \text{cone} \qquad 6.5.2\text{-}2$$

The justification and basis for these transition criteria are discussed in detail in References 6.5-1 and 6.5-2. In using this criteria, it is presumed that an instantaneous transition from laminar to turbulent flow occurs. In practice, transition occurs gradually.

6.5.2.1 Development of Approximate Transition Relationship - From the definition of momentum thickness, Reynolds number, and local Mach number, R_{e_θ}/M_L, it can be shown that

$$\rho_\infty \propto \frac{(R_{e\theta}/M_L)^2}{X\ \tan\ \delta_{EFF}} \qquad 6.5.2\text{-}3$$

This relationship indicates that the free stream density (or altitude) depends primarily on the ($R_{e\theta}/M_L$) parameter, local deflection angle and reference distance - not on free stream velocity. The weak dependence of transition altitude on velocity is illustrated in Figure 6.5-2.

The X tan δ_{EFF} parameter of Equation 6.5.2-3 was utilized to correlate the transition data calculated by Equations 6.5.2-1 and 6.5.2-2. Better correlation was obtained by plotting $\rho\infty$ vs($X \tan^{1.5} \delta_{EFF} + .2$) rather than ($X \tan \delta_{EFF}$) as illustrated in Figures 6.5-3 and 6.5-4.

The transition equations for a wedge ($R_{e\theta}/M_L = 150$) derived from Figure 6.5-4 is:

$$(\rho\infty)_{TR} = 7.2 \times 10^{-6}\psi^{-1.234} \qquad \text{For } \psi > .75 \qquad 6.5.2\text{-}4$$

$$\text{or } (\rho\infty)_{TR} = 1.0 \times 10^{-5} \qquad \text{For } \psi \leq .75 \qquad 6.5.2\text{-}5$$

$$\text{where } \psi = X \tan^{1.5} \delta_{EFF} + .2 \qquad 6.5.2\text{-}6$$

Similarly, for a cone ($R_{e\theta}/M_L = 200$) as shown in Figure 6.5-4 is

$$(\rho\infty)_{TR} = 1.1 \times 10^{-5}\psi^{-1.1284} \qquad \text{For } \psi > .75 \qquad 6.5.2\text{-}7$$

$$\text{or } (\rho\infty)_{TR} = 1.6 \times 10^{-5} \qquad \text{For } \psi \leq .75 \qquad 6.5.2\text{-}8$$

6.5.2.2 Transition Logic - Four choices of transition logic are available and are identifiable by the index number:

Index Number	Transition Characteristics
0	Always turbulent flow
1	Always laminar flow
2	Calculate transition using flat plate
3	Calculate transition using cone

In addition, the logic assumes that if transition occurs in the forward portion of the vehicle all aft points, side, top, fins, and leading edges experience transition at the same time (except when laminar flow is specified, index number "1"). The transition criteria outlined above yields realistic maximum heating rates and temperatures.

FIGURE 6.5-2

EFFECT OF VELOCITY ON TRANSITION ALTITUDE OF A CONE

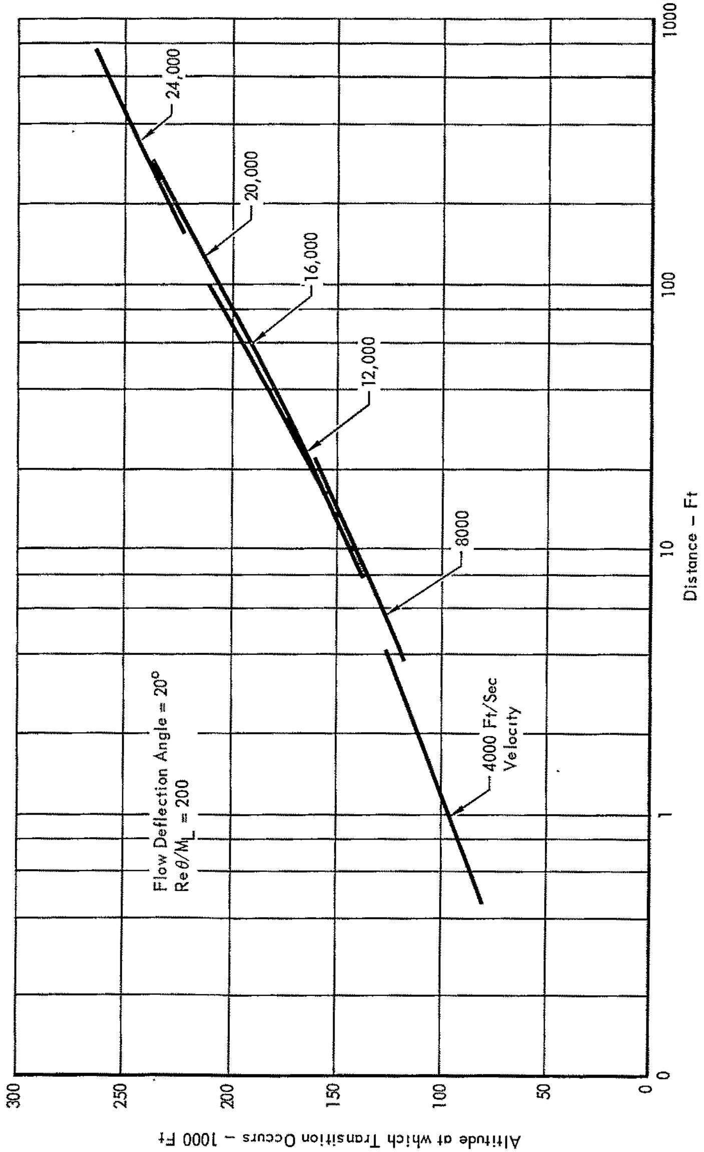


FIGURE 6.5-3

TRANSITION CORRELATION FOR FLAT PLATE AT $\frac{RE\theta}{ML} = 150$

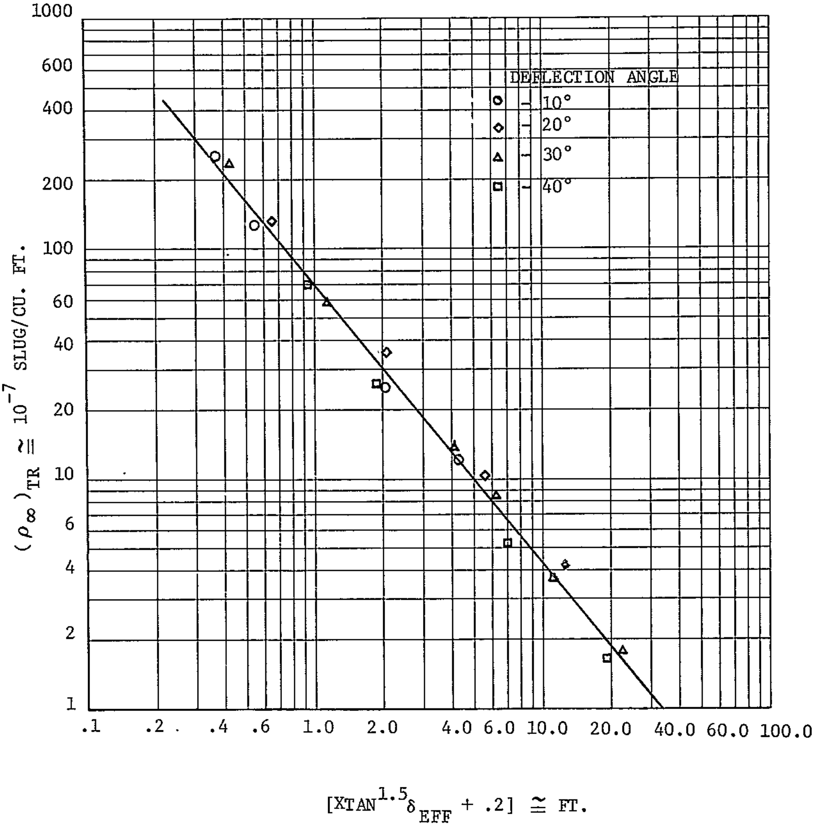
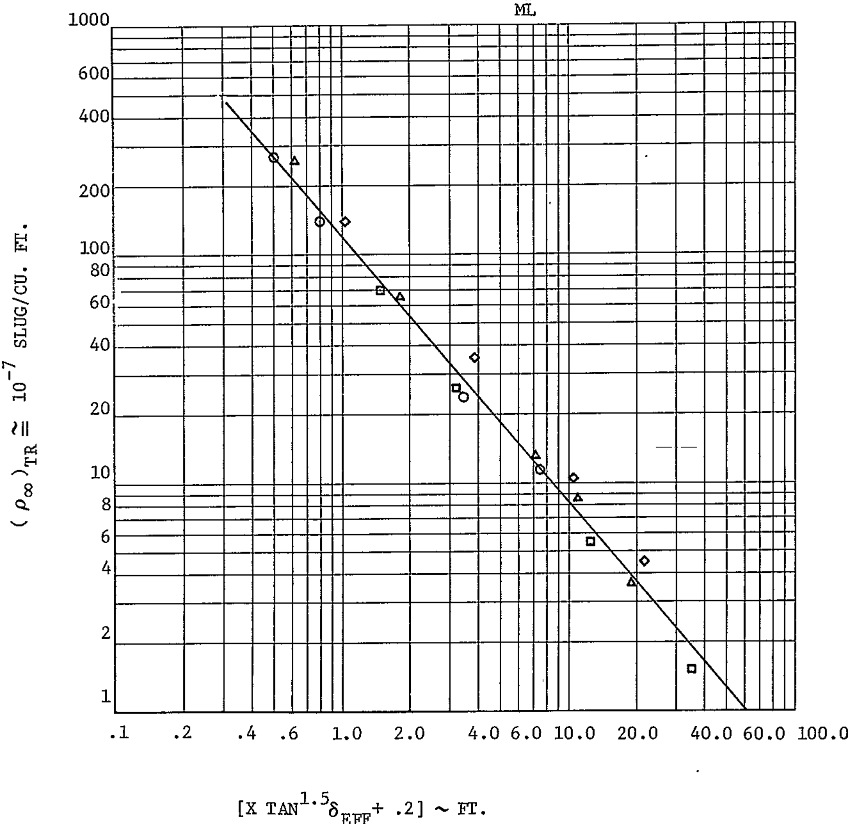


FIGURE 6.5-4

TRANSITION CORRELATION FOR CONE AT $\frac{R_{E\Theta}}{ML} = 200$



6.5.3 Laminar Heating Equations - A number of rigorous analytical theories have been developed to predict the magnitude of heating associated with laminar boundary layers. Laminar heat transfer theories can be classified according to the vehicle geometry to which they apply such as nose cap, leading edge, flat plate, cone, etc. Available computer programs are considered too sophisticated and time consuming for rapid calculations. The approach followed herein is to start with the rigorous analytical method, reduce them to their simplest form by neglecting second order effects, and then adjust the approximate methods to yield the best possible correlation with analytical data.

The approximate relationships developed by Hankey-Neumann (Reference 6.5-3) are used to estimate flat plate heat transfer rates. In these equations, the heat transfer rate is explicitly related to free stream density, velocity, local flow deflection angle, and reference length. Thus, computer solution is very rapid.

Equations are available in TEMP for calculating heat transfer rates for nose caps (stagnation point), flat plates, cone, leading edges, upper surface with separate flow, or any other type of surface where the local to stagnation point heat flux ratio (ratio method) is known. At any of the 27 body points, these equations are called for by specifying the equation number in the input data matrix (DTP). The laminar heat flux equations are used in the program when the boundary layer flow is determined to be laminar by the transition criteria or the flow is specified laminar in the input data matrix.

6.5.3.1 Stagnation Point Heating (Or Constant Ratio Method)

$$\dot{q}_{cw} = \left[j_L\right] \frac{4.46 \times 10^{-9}}{\sqrt{R}} \left[\rho_\infty\right]^{1/2} \left[V_\infty\right]^{3.15} \qquad 6.5.3\text{-}1$$

Equation 6.5.3-1 is the well known Detra, Kemp and Riddell empirical relationship for estimating stagnation point heat transfer rates. Equation 6.5.3-1 can also be used to determine body heat transfer rates by letting the j_L-factor be the ratio of local to stagnation point heat fluxes. For ballistic vehicles, the nose radius obtained from the geometry subroutine is the radius of curvature of the blunt heatshield, however, reference lengths are measured from the small end of the vehicle. It is recommended that for ballistic vehicles, all body heating rates be computed by specifying ratios of local to stagnation point heating. For lifting vehicles, the nose radius is the radius of the small end. The free stream density and velocity are inputs from the aero subroutine.

6.5.3.2 Ratio Method with Deflection Angle Variation - If the ratio of local to stagnation point heat flux is to be used and the angle-of-attack varied during entry, the following equation, which includes the effect of variable flow deflection angle, should be used rather than Equation 6.5.3-1.

$$\dot{q}_{cw} = \frac{4.46 \times 10^{-9}}{\sqrt{R}} \left[\rho_\infty\right]^{1/2} \left[V_\infty\right]^{3.15} \left[F(\delta_{EFF})\right]_L \left[j_L\right] \qquad 6.5.3\text{-}2$$

where from Reference 6.5-3 in modified form:

$$\left[F(\delta_{EFF})\right]_L = \frac{\sin(\delta_{EFF})\cos^{1/2}(\delta_{EFF}) + .08}{\left[.186 + .314 \sin^2(\delta_{EFF})\right]^{.25}} \qquad 6.5.3\text{-}3$$

Note when inputting a value of j_L for this equation, the product of $j_L[F(\delta_{EFF})_{REF}]$ represents the local to stagnation point heat flux ratio at the specified flow deflection angle. Also note that if δ_{EFF} is negative the program sets δ_{EFF} equal to 0.

6.5.3.3 Swept Cylinder (Leading Edge) - The swept cylinder theory of Reference 6.5-4 which reduces the stagnation point heat flux according to the magnitude of sweep, is employed for leading edge calculations.

$$\dot{q}_{cw} = \frac{4.46 \times 10^{-9}}{\sqrt{2}\, R_{LE}} \left[\rho_\infty\right]^{1/2} \left[V_\infty\right]^{3.15} \left[j_L\right] \times \left[\cos^{1.7}(\lambda_{EFF}) + .26 \sin^2(\lambda_{EFF})\right] \qquad 6.5.3\text{-}4$$

6.5.3.4 Flat Plate and Cone - For flat plate laminar heat flux calculation, the simplified equation developed by Hankey-Neumann, Reference 6.5-3, in slightly modified form, is utilized:

$$\dot{q}_{cw} = \frac{1.695 \times 10^{-9}}{\sqrt{X}} \left[j\right] \left[\rho_\infty\right]^{1/2} \left[V_\infty\right]^{3.15} \left[F(\delta_{EFF})\right]_L \left[1 + 2\frac{H_w}{H_s}\right] \qquad 6.5.3\text{-}5$$

where: $[F(\delta_{EFF})]_L$ is defined by Equation 6.5.3-3.

To eliminate the need of iteration, the following approximation was utilized to obtain the wall enthalpy (H_w) explicitly:

$$\left[1 + 2\frac{H_w}{H_s}\right] = \left\{\left(1 + \frac{3.014 \times 10^{+7}}{V_\infty^2 + 5.007 \times 10^6}\right) \times \left[\frac{\dot{q}_1}{\varepsilon}\left(1 + \frac{3.014 * 10^7}{V_\infty^2 + 5.007 * 10^6}\left(\frac{\dot{q}_1}{\epsilon}\right)^{1/4}\right)\right]^{1/4}\right\} \qquad 6.5.3\text{-}6$$

where $$\dot{q}_1 = \dot{q}_{cw} \Big/ \left(1 + 2\frac{H_w}{H_s}\right) \qquad 6.5.3\text{-}7$$

(This approximation was derived by the technique described in Section 6.5.5.1.)

The original Hankey-Neumann equations for flat plate heating were modified in order to obtain a better correlation with data predicted with Eckert's Reference Enthalpy method. The degree of correlation is illustrated in Figure 6.5-5. Excellent agreement is obtained by the empirical method considering the wide range in deflection angle (-5° to 40°), altitude (150,000 to 250,000 ft.) and velocity (12,000 to 23,000 ft/sec).

Equation 6.5.3-5 can be employed for conical surface heating by setting the j_L factor = $\sqrt{3}$, as determined from the Mangler transformation between cone and flat plate heating.

6.5.3.5 Separated Flow - For top surfaces at rather large angles-of-attack the flow will become separated; thus, the heat flux will become insensitive to the flow deflection angle. The following relationship was obtained from correlation of ASSET wind tunnel and flight data for heating with separated flow:

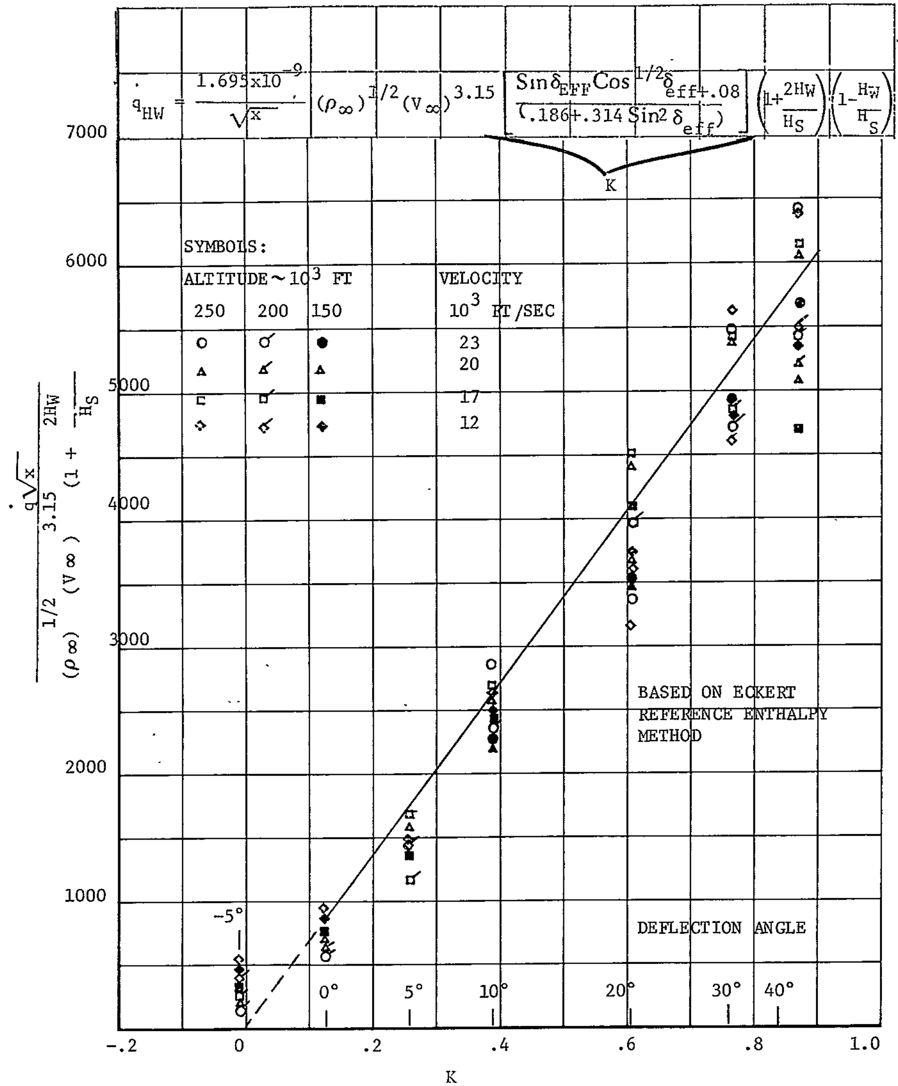
$$\dot{q}_{cw} = 3.65 \times 10^{-7}\, \rho_\infty V_\infty^{\,3} \qquad 6.5.3\text{-}8$$

The user specifies which areas have separated flow for the entire trajectory.

6.5.4 Turbulent Heating Equations - Turbulent heat transfer equations were developed in a manner similar to the laminar equations discussed in Section 6.5.3.

FIGURE 6.5-5

LAMINAR FLAT PLATE HEATING APPROXIMATION EQUATION



6.5.4.1 Sonic Line or Constant Ratio Method - The sonic line heating on a hemisphere is used as a reference value for turbulent flow calculations. The following relationship was derived from correlation of sonic line heat transfer data (van Driest Theory) reported in Reference 6.5-5.

$$\dot{q}_{cw} = [j_T] \frac{9.36 \times 10^{-9}}{R^{.2}} [\rho\infty]^{.78} [V\infty]^{3.45} \qquad 6.5.4\text{-}1$$

Equation 6.5.4-1 can be utilized to predict turbulent heat transfer rates at other body points by setting the (j_T) factor equal to the ratio of local to sonic line heat fluxes.

6.5.4.2 Ratio Method with Deflection Angle Variation - The following equation should be used rather than Equation 6.5.4-1 if the angle-of-attack varies during entry.

$$\dot{q}_{cw} = [j_T] \frac{9.36 \times 10^{-9}}{R^{.2}} [\rho\infty]^{.78} [V\infty]^{3.45} [F(\delta_{EFF})]_T \qquad 6.5.4\text{-}2$$

where from Reference 6.5-1 in modified form:

$$[F(\delta_{EFF})]_T = \frac{\cos^{.8}(\delta_{EFF}) \sin^{1.6}(\delta_{EFF}) + .01}{[.198 + .302 \sin^2(\delta_{EFF})]^{.7}} \qquad 6.5.4\text{-}3$$

Note when inputting a value for j_T the product of j_T $[F(\delta)_{REF}]$ represents the local to sonic line heat flux ratio at the specified deflection angle (See Section 6.5.3.2).

6.5.4.3 Flat Plate and Cone - Two flat plate heat flux relationships are utilized to cover the entire range of free stream velocities:

$$\dot{q}_{cw} = [j_T] \frac{1.35 \times 10^{-8}}{X^{.2}} [\rho\infty]^{.78} [V\infty]^{3.45} [F(\delta_{EFF})]_T \text{ for } V\infty \geq 8000 \text{ ft/sec} \qquad 6.5.4\text{-}4$$

$$\dot{q}_{hw} = [j_T] \frac{9.1 \times 10^{-9}}{X^{.2}} [\rho\infty]^{.40} [V\infty]^{2.91} [F(\delta_{EFF})]_T \text{ for } V\infty < 8000 \text{ ft/sec} \qquad 6.5.4\text{-}5$$

where $[F(\delta_{EFF})]_T$ is defined by Equation 6.5.4-3.

Equation 6.5.4-4 is basically the Hankey-Neumann equation from Reference 6.5-1 with the exponents changed to yield better correlation with turbulent heat transfer data predicted with the Eckert's Reference Enthalpy method. The degree of data correlation provided with Equation 6.5.4-4 is illustrated in Figure 6.5-6. Note that the higher velocity equation yields a cold wall heating rate which is converted to a hot wall value with Equation 6.5.5-1, whereas Equation 6.5.4-5 yields hot wall value directly. Setting $j_T = \sqrt[5]{2}$ in Equations 6.5.4-4 and 6.5.4-5 allows calculation of turbulent heating for conical surfaces.

6.5.5 Additional Heating Parameters - The remaining steps in the temperature subroutine are determination of the hot wall heating rate, conversion of heating rate to equilibrium temperature and calculation of the maximum equilibrium temperature, heating duration and total heat.

6.5.5.1 Hot Wall Heating Rate Correction - All of the laminar and turbulent heat flux rates (except the low speed turbulent values) are based on a cold wall temperature. In reality the surface temperature will be higher, thus, the heat flux to the surface will be less. The reduction of cold wall heating rate due to higher surface temperatures is given by:

$$\dot{q}_{HW} = \dot{q}_{CW}\left(1 - \frac{H_w}{H_s}\right) \qquad 6.5.5\text{-}1$$

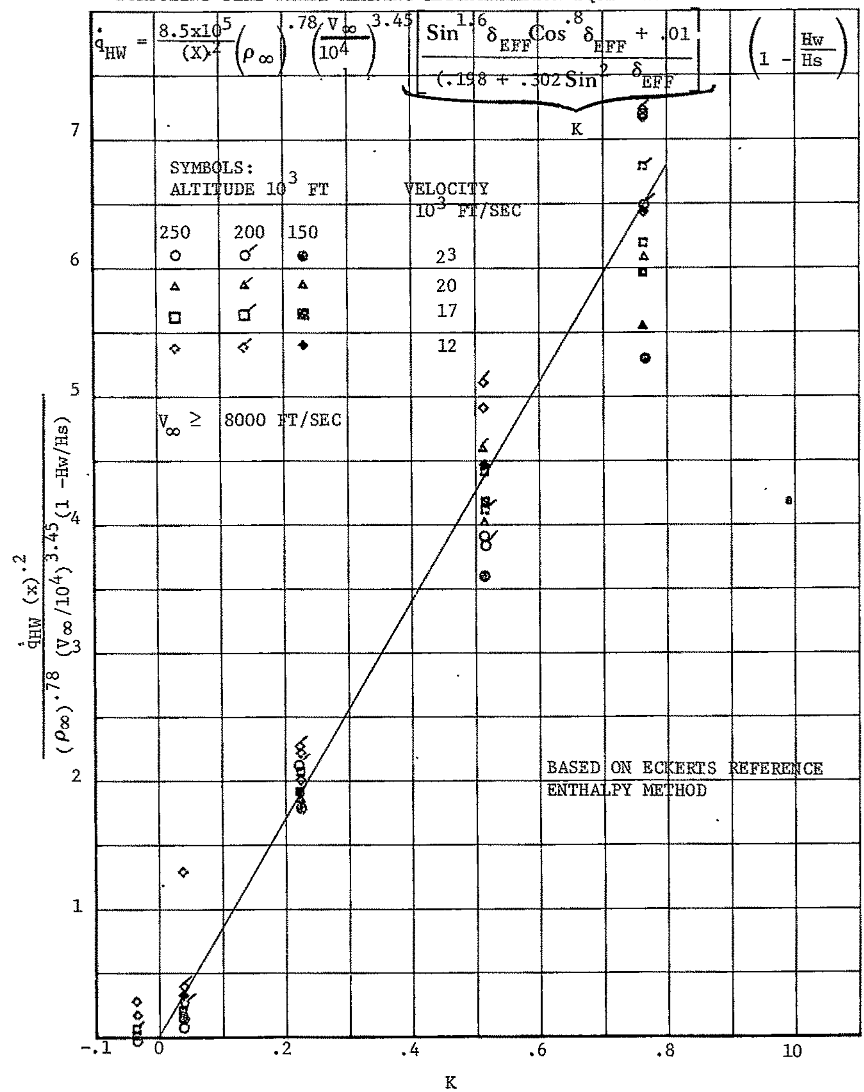
To eliminate iteration for the wall enthalpy (H_w), the following approximation was developed for H_w:

$$\text{Let } \frac{H_w}{H_s} = \frac{C_p T_w}{\frac{V_\infty^2}{2\,J_g} + H_\infty} = 2\,JgC_p\,\frac{\left(\frac{\dot{q}_{CW}}{\sigma\varepsilon}\right)^{1/4}}{V_\infty^2 + 2JgH_\infty} = \frac{C_1\left(\frac{\dot{q}_{CW}}{\varepsilon}\right)^{1/4}}{V_\infty^2 + 5.0007 \times 10^4 \times 100} = \frac{C_1\left(\frac{\dot{q}_{CW}}{\varepsilon}\right)^{1/4}}{V_\infty^2 + 5.007\times10^6} \qquad 6.5.5\text{-}2$$

$$\text{Where: } C_1 = \frac{Cp2Jg}{(\sigma)^{1/4}} = \frac{.25 \times 5.007 \times 10^4}{[.4758 \times 10^{-12}]^{1/4}} = 1.507 \times 10^7$$

FIGURE 6.5-6

TURBULENT FLAT PLATE HEATING APPROXIMATION EQUATION



1st Approximation (Substituting 6.5.5-2 into 6.5.5-1)

$$(\dot{q}_{HW})_1 = \dot{q}_{CW} \left[1 - \left(\frac{1.507 \times 10^7 \left(\frac{\dot{q}_{CW}}{\varepsilon} \right)^{1/4}}{V_\infty^2 + 5.007 \times 10^6} \right) \right] \qquad 6.5.5\text{-}3$$

2nd Approximation (Substituting $(\dot{q}_{HW})_1$ for $\dot{q}_{CW}$ in 6.5.5-2 and $\frac{H_w}{H_s}$ into 6.5.5-1)

$$\dot{q}_{HW} = \dot{q}_{CW} \left\{ 1 - \frac{1.507 \times 10^7}{V_\infty^2 + 5.007 \times 10^6} \left[\frac{\dot{q}_{CW}}{\varepsilon} \left[1 - \frac{1.507 \times 10^7}{V_\infty^2 + 5.007 \times 10^6} \left(\frac{\dot{q}_{CW}}{\varepsilon} \right)^{1/4} \right] \right]^{1/4} \right\} \qquad 6.5.5\text{-}4$$

Since the hot wall correction factor converges rapidly, two successive approximations provide sufficient accuracy.

6.5.5.2 Equilibrium Surface Temperature - The equilibrium temperature is simply the temperature attained by a surface when it radiates all of the incident aerodynamic heating.

$$T_{HW} = 1204.5 \left(\frac{\dot{q}_{HW}}{\varepsilon} \right)^{1/4} - 460. \qquad 6.5.5\text{-}5$$

6.5.5.3 Maximum Surface Temperature - The peak temperature for the entire trajectory is determined by comparing each point in the trajectory with the previous time-step calculation, accepting the larger values and disregarding the lower numbers.

$$\text{IF } (T_{HW})_i > (T_{HW})_{i-1} \qquad 6.5.5\text{-}6$$

$$T_{MAX} = (T_{HW})_i \qquad 6.5.5\text{-}7$$

6.5.5.4 Heating Time - The heating time is the duration of heating above some specified value (e.g., 1.0 BTU/FT^2 - SEC).

when $\dot{q}_{HW} \geq C$

$$Q_{TIME} = \sum_{i=1}^{n} (\tau_i - \tau_{i-1}) - \tau_{init.} \qquad 6.5.5\text{-}8$$

6.5.5.5 Total Heat - The total heat is simply the integration of the hot wall heat flux-time history and is accomplished as follows:

$$Q_T = \sum_{i=1}^{n} \left(\frac{[(\dot{q}_{HW})_i + (\dot{q}_{HW})_{i-1}]}{2} (\tau_i - \tau_{i-1}) \right). \qquad 6.5.5\text{-}9$$

The peak temperature (T_{MAX}), total heat (Q_T) and heating duration (Q_{TIME}) serve as inputs to the Thermal Protection Subroutine and are used to calculate the thermal protection weights for each of 27 areas defining the entry vehicle.

6.5.6 Flow Diagram - The logic flow of this subroutine is shown in Figure 6.5-7.

6.6 Thermal Protection System Subroutine - Hypersonic spacecraft require a reliable and efficient Thermal Protection System (TPS) in order to dissipate the heat loads encountered during entry. In general, the TPS is comprised of either ablative, ceramic, or radiative metal shields backed with low density fiberous insulation with or without an active structural cooling system.

In developing the TPS subroutine, the complexity or sophistication of the subroutine was minimized and sufficient flexibility was provided to permit handling of a variety of thermal protection concepts. Figure 6.6-1 depicts the types of thermal protection designs available. The load carrying structure below the hard insulation block is not included in TPS, but is included in another subroutine called STRUCT.

The entry thermal environment, is calculated by a combination of AERO, TEMP & GEOM subroutines. Two trajectories can be employed to define the most critical combination of peak temperature, heat load and heating duration for the mission flight envelope. The spacecraft surface area is subdivided into 27 areas, comprised of a nose cap, 12 body sections, 6 fin surfaces, 6 fin leading edges, and 2 body leading edges.

The choice of radiative or ablative-ceramic heat shielding is made on the basis of the peak allowable metal shingle temperature which is an input matrix variable for each of the 27 sections.

FIGURE 6.5-7

TEMP SUBROUTINE FLOW CHART FOR TYPICAL SURFACE LOCATION

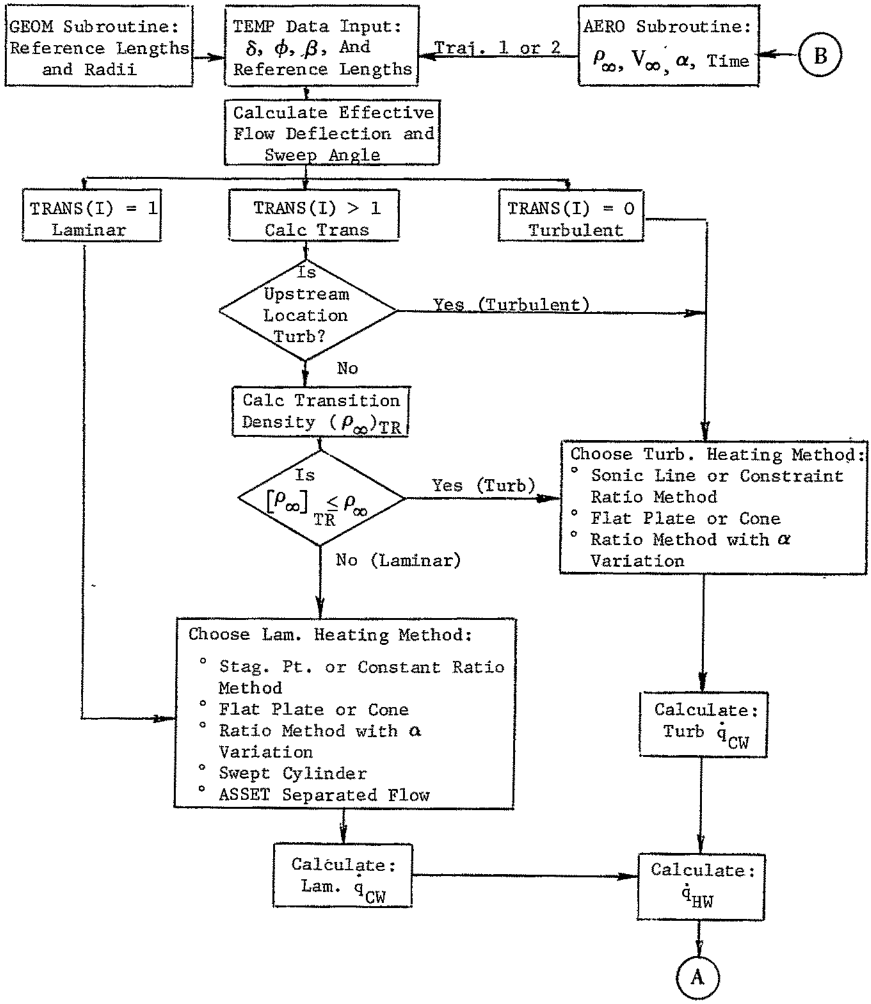


FIGURE 6.5-7
(Cont'd)

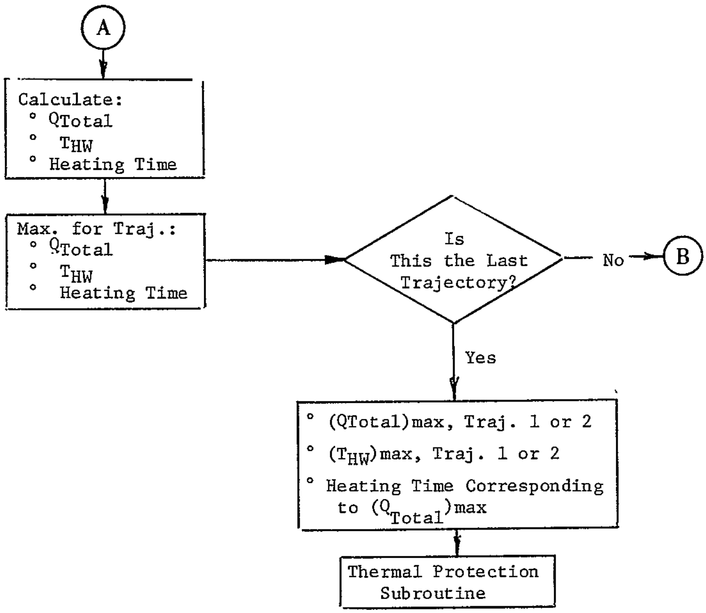
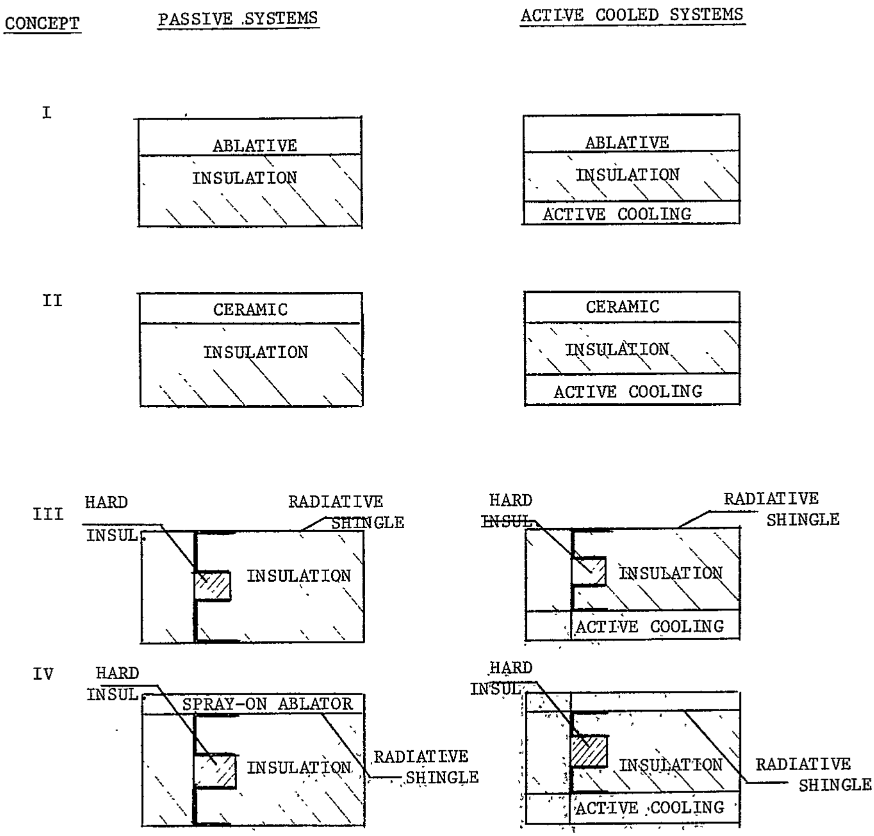


FIGURE 6.6-1

THERMAL PROTECTION CONCEPTS CAPABLE OF ANALYSIS BY TPS



TPS automatically selects a metal and weighs a radiative shingle if the calculated temperatures are below the maximum allowable metal temperature. If allowable temperatures are exceeded, TPS uses input material performance parameters to weigh an ablative or ceramic panel. After the outer panel or shingle is sized, TPS sizes insulation and determines the inner wall cooling system weight. A safety factor of 1.07 is applied to the temperature (in °R) calculated in the TEMP subroutine before it is compared to the maximum allowable shingle temperature. If the fin surfaces are less than 1600°F the shingle weights are calculated in the STRUCT subroutine instead of TPS. Notice also that load carrying hard insulation is included only for the 13 body surfaces.

6.6.1 Ablator Material - The ablative heat shield is sized to restrict the backside temperature to prescribed limits for a given heat load. The heat load is defined by the total heat and heating times obtained from the TEMP subroutine. The heat shield thickness (or unit weight) required to maintain a given maximum bond line temperature is estimated with the semi-empirical relationship

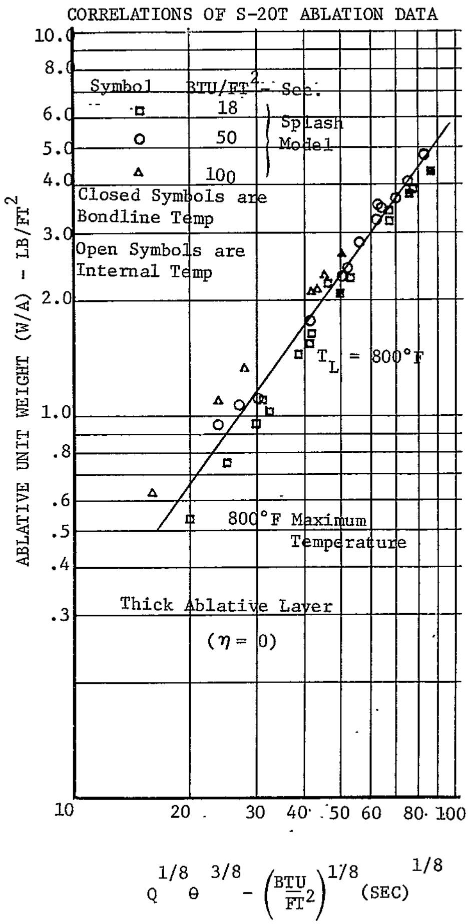
$$W/A = SL\emptyset \; (Q^{1/8}\theta^{3/8})^{EXP} \qquad 6.6.1\text{-}1$$

where: SLØ and EXP are constants, dependent on the material and maximum bond line temperature. A, Q (total heat), and θ (heating time) are supplied by TEMP at each of 27 body points.

The adequacy of the semi-empirical approach is proven by correlating ablation arc data for three elastromeric materials tested over a wide range of heating conditions; and by showing good agreement with heat shield weights predicted by computerized ablation analysis. Figure 6.6-2 typifies the degree of plasma-arc data correlation obtained with Equation 6.6.1-1 for a bondline temperature of 800°F.

Equations, similar to Equation 6.6.1-1, are available in TPS to determine ablative weights depending on the input code ABLAT;

FIGURE 6.6-2


CORRELATIONS OF S-20T ABLATION DATA
Symbol
BTU/FT² - Sec.
18
50
100
Splash Model
Closed Symbols are Bondline Temp
Open Symbols are Internal Temp
T_L = 800°F
800°F Maximum Temperature
Thick Ablative Layer
(η = 0)
ABLATIVE UNIT WEIGHT (W/A) - LB/FT²
10.0
8.0
6.0
5.0
4.0
3.0
2.0
1.0
.8
.6
.5
.4
.3
10
20
30
40
50
60
80
100
$Q^{1/8}\ \theta^{3/8}$ - $\left(\frac{BTU}{FT^2}\right)^{1/8}$ $(SEC)^{1/8}$

CODE: (1) WABL(I) = [SLØ 1* QTHETA(I) ** EXP1] * AREA (I) 6.6.1-2

(2) WABL(I) = [SLØ 2* QTHETA(I) ** EXP2] * AREA (I) 6.6.1-3

(3) WABL(I) = [SLØ 3* QTHETA(I) ** EXP3] CØN 3 * AREA (I) 6.6.1-4

Where: QTHETA = $Q^{1/8}\theta^{3/8}$

The user specifies the equation code number and supplies the proper SLØ, EXP or CØN corresponding to the material and bond line temperature desired. Thus, a combination of three different ablative materials or maximum bond line temperature options are available. The area of each section is provided by the GEOM subroutine.

6.6.1.1 Back-Up Structure - The ablative heat shield panel consists of a layer of reinforced elastomeric ablative material, bonded to a removable stiffened metal panel. Metal panels are assumed to be constructed from a titanium skin, stiffened with longitudinal corrugations. Four centrally located screws, symmetrically spaced five inches from the edge of the panel, attach each panel, through insulated clips, to the body rings. Hard insulation pads between ring and panel provide a bearing surface for panel pressure loads. Straps attached to inside crests of corrugations along ring bearing lines provide sufficient orthogonal strength to react outward pressures on each panel. Figure 6.6-3 shows the relationship of the panel system components. Back-up structure weight is computed by;

WABLS(I) = .55 * AREA (I) * ABLS +.13 * AREA(I) 6.6.1-5

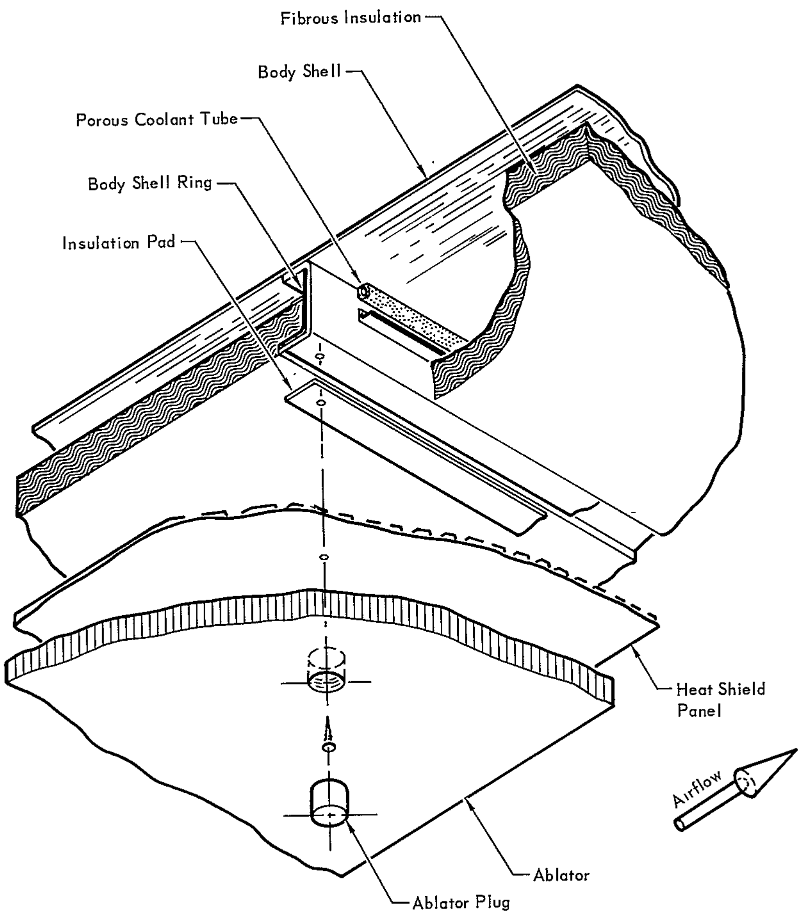
where .55 * AREA(I) is metal panel weight (Ti)
ABLS is an input multiplier
.13 * AREA(I) is the glue needed for bonding

6.6.1.2 Spray-On Ablator - In general, high altitude aborts or very steep entries are characterized by high heat flux spikes exceeding the temperature limits of radiative shingle design. If the radiative approach is desired, then the metallic shingles must be protected by a layer of ablative material applied by either spraying or adhesively bonding pre-formed sheets.

The spray-on ablative weight is calculated by specifying either of the following equations using the code word ABLAT;

FIGURE 6.6-3

ABLATIVE HEAT SHIELD PANEL



CODE: (6) WABL(I) = SLØ 6 * (TMAX(I) - TIMT 6) ** EXP 6 +CØN*AREA(I) 6.6.1-6

(7) WABL(I) = SLØ 6 * QTHETA(I) ** EXP 6 +CØN 6 * AREA(I) 6.6.1-7

where: TIMT6 is the inputed temperature limit of the radiative shingles and QTHETA = $Q^{1/8}\theta^{3/8}$

The weight of radiative shingle is then calculated on the basis of TIMT6 rather than TMAX(I).

6.6.2 Ceramic - Ceramic heat shields are normally used in regions that are exposed to high heating rates and shear loads such as nose tip or leading edge areas. There are two ways to compute ceramic weights in TPS and they are selected by the value input for the code word ABLAT. The first is an empirically derived correlation based on the ASSET and Dynasoar data shown in Figure 6.6-4.

CODE: (4) WCERM(I) = .185 * TMAX(I) ** .38 * AREA(I) 6.6.2-1

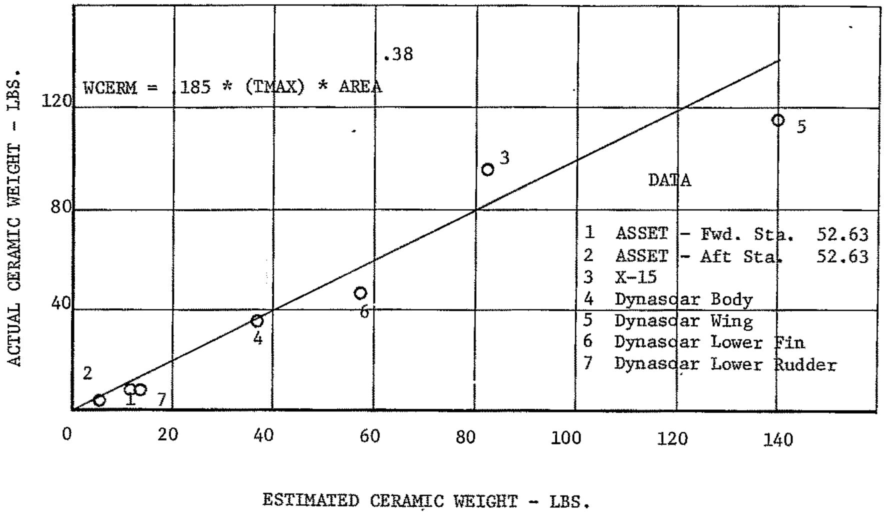
This equation is based on enclosed surface area rather than exposed (wetted) area. This equation can be used only for leading edge sections where enclosed area is obtained from the GEOM subroutine. The other equation is based on the approximations derived in Reference 6.6-1 and is:

CODE: (5) $$\text{WCERM(I)} = \text{SLØ5} * \left\{\frac{5.602 \times 10^{-5} \frac{Ke\theta}{C_p}}{.214 - \ln\left[1 - \frac{\text{TMAX5} - \text{Ti}}{\text{Ts} - \text{Ti}}\right]}\right\}^{1/2} * \text{AREA(I)} \qquad 6.6.2\text{-}2$$

6.6.3 Radiative Shingle - Radiative metal shingle TPS is usually lighter but more expensive per unit than ablative TPS if temperatures are below the maximum allowable metal temperature. If the allowable temperature is low enough, the metal shingle can be reused and unit costs greatly reduced. Our hardware experience in the Gemini and ASSET programs and results of in-house studies has led to the radiative shingle design shown in Figure 6.6-5. The TPS subroutine sizes all components shown except the support frames and the stiffened shell which are sized in the STRUCT subroutine.

COMPARISON OF CERAMIC NOSE CAP AND LEADING EDGE WEIGHTS

FIGURE 6.6-4



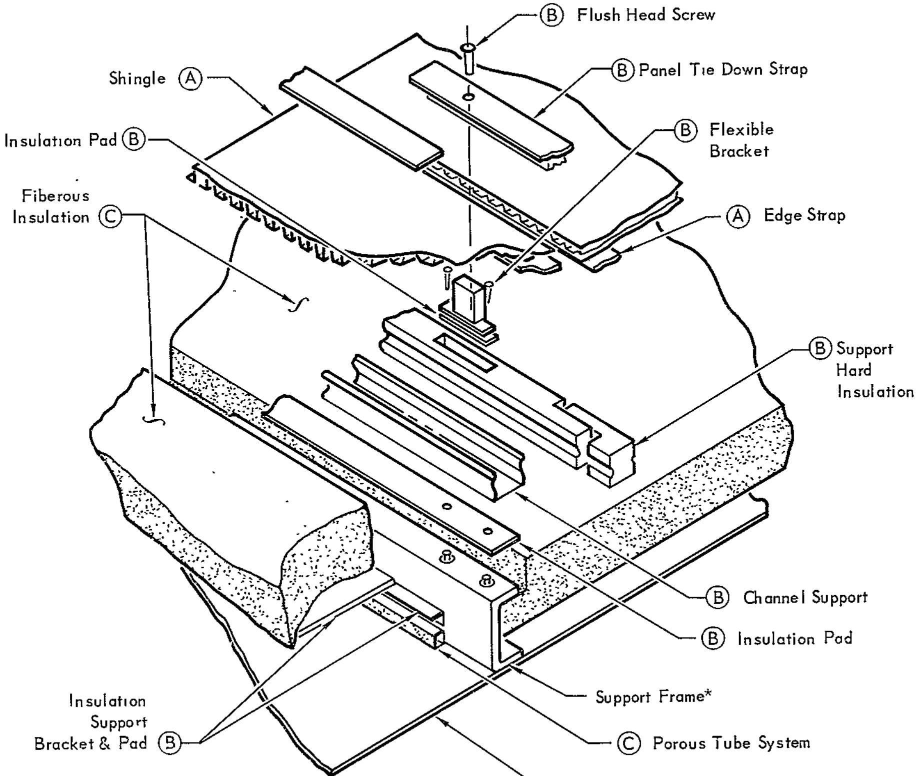
6.6.3.1 Basic Shingle - The shingle panel is a single skin open faced corrugated sandwich. The panel geometry was optimized so that the design was equally critical in tension and compression for applied bending moments. The sandwich elements were restricted to slenderness ratios, b/t, of less than 120 in order to avoid exceedingly long unsupported elements that could easily be damaged during handling.

A design panel pressure of 2 psi was selected as a conservative estimate of the reentry pressure. Most high L/D reentry trajectories have considerably lower pressures at peak temperature. However, since off the pad abort results in high pressures at low temperatures and in order to limit the number of conditions that need be checked, the above 2 psi pressure at peak temperature was selected. The unit shingle weights and back-up structure computed by TPS are shown in Figure 6.6-6.

Applicable materials for specific temperature ranges are noted. Although Figure 6.6-6 indicates the superior weight advantage of molybdenum in comparison to columbium above 2540°F, the present MDAC policy is to use columbium up to its maximum temperature. At this maximum temperature material

FIGURE 6.6-5

THERMAL PROTECTION SYSTEM CONCEPT
(RADIATIVE)



*These items are not included in the thermal protection unit weights selection is changed to molybdenum and unit weight is held constant until the molybdenum unit weight curve, which is lighter than columbium, is intersected. Above this intersection temperature, weights used are those shown on the molybdenum curve. Maximum temperature use limit referred to above is dependent upon whether single flight or multiple reuse is required.

Although the above is the current program policy, four optional sets of TPS equations are available. These sets of equations, derived from Figure 6.6-6, allow the option of selecting the most reliable or minimum weight system, each with the option of reuse or only one flight depending on a selector switch called WHICH.

6.6.3.2 Hard Insulation - This insulation block functions as a thermal resistor between the straps supporting the high temperature shingles and the cooler load carrying structural rings (see Figure 6.6-5).

6.6.4 Inner Wall Cooling - Low density insulation, with or without an active cooling subsystem, is normally employed in most thermal protection concepts to maintain the load carrying structure temperatures at acceptable limits. The type of structural cooling system is selected in TPS by specifying the equation number (code name SULAT).

Passive Insulation Design - This mode relies only on low density insulation to reduce the high surface temperatures. The method of sizing the insulative requirements is based on the analysis in Reference 6.6-1. Unit weight for insulation behind surface shingles is:

$$W/A = \left[\frac{5.602 \times 10^{-5} \;*\; \dfrac{.473}{\eta + .473} \;*\; \dfrac{K\rho\theta}{C_p}}{.214 - LN \left(1 - \dfrac{TMAX2 - Ti}{TSURF - Ti}\right)} \right]^{1/2} \qquad 6.6.4\text{-}1$$

where: K = Thermal conductivity

C_p = Specific Heat

η = Heat capacitance ratio between backstructure and insulation

$$\left(\rho C_p \delta\right)_2 \Big/ \left(\rho C_p L\right)_1$$

STRUCTURAL HEAT PROTECTION SYSTEM UNIT WEIGHTS

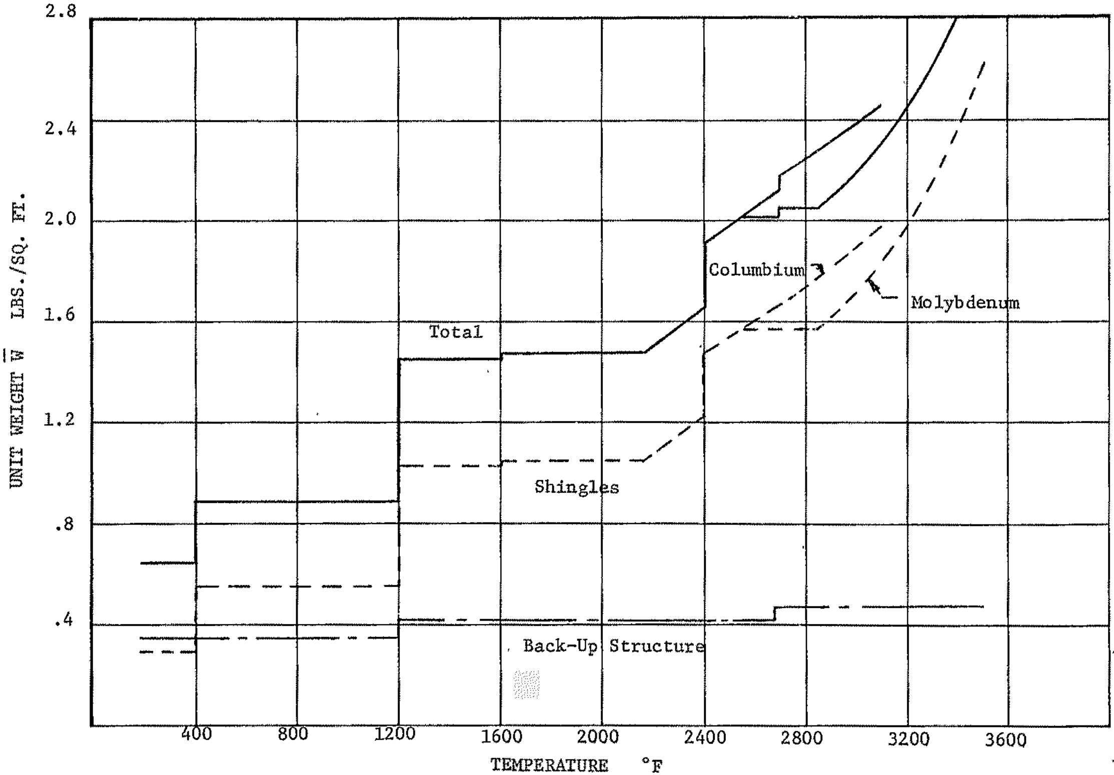


FIGURE 6.6-6

Ti = Initial temperature

TMAX1 = Bond line temperature

TMAX2 = Maximum backside temperature limit

$$TSURF = \left(\frac{QHW}{QTIME}\right)^{1/4} \text{ Radiative surface}$$

$$= \left(2/3\right)\left(TMAX1\right) - \text{ Ablative surface}$$

Since the hard spacer is an insulator with an imposed surface heating condition and a maximum backface temperature limit, the analysis is exactly the same and the unit weight is;

$$W/A = \left[\frac{5.602 \times 10^{-5} * \frac{.473}{\eta + .473} * \frac{K\rho\theta}{C_p}}{.214 - LN\left(1 - \frac{TMAX2 - Ti}{TSURF - Ti}\right)}\right]^{1/2} \qquad 6.6.4\text{-}2$$

Conductivity is evaluated at the average temperature;

$$K = SL\emptyset 9 \frac{(TSURF + TMAX1 + 920)}{2} C\emptyset NK9$$

The spacer thickness is simply

$$RSIL = \frac{12 W/A}{\rho} \qquad 6.6.4\text{-}3$$

and the weight of hard insulation is

WSIAV = HIAREA * AREA (I) * WRADS(I) 6.6.4-4

where: HIAREA is the ratio of hard insulation area per square foot of shingle area.

Because the conductivity (K) of low density insulation varies appreciably with temperature and pressure, its value for use in Equation 6.6.4-4 is determined as a linear function of average temperature;

$$K = SI\emptyset K \text{ Tave } (°R) + C\emptyset NK \qquad 6.6.4\text{-}5$$

where: $Tave = \frac{TSURF + TMAX2}{2}$ radiative design

or $Tave = \frac{2/3 \; TMAX1 + TMAX2}{2}$ ablative design

SLØK, CØNK are the conductivity slope-intercept values

Based on past experience, a reasonable average pressure for evaluating thermal conductivity is at an equivalent altitude of 150,000 feet (1mmHg).

In Figure 6.6-7, this approximate method of predicting insulation requirements is shown to compare extremely well with the exact computer solutions of a hypothetical insulative design exposed to a two-step heat flux profile. In general, the TPS estimated insulation weights will be conservative since an adiabatic backface, i.e., no heat flow into the cabin, was presumed in developing Equation 6.6.4-1. The weight of the passive system per panel is computed by one of the following equations;

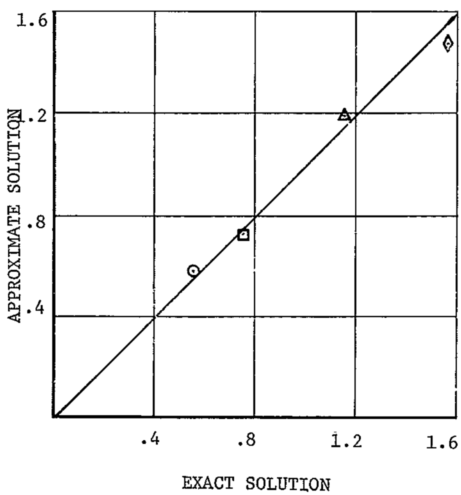
WINS = SØL 10 * W/A * AREA 6.6.4-6

WINS = SLØ 11 * W/A * AREA 6.6.4-7

WINS = CØN 12 * AREA 6.6.4-8

6.6.4.1 <u>Active Structural Cooling System</u> - It has been shown in a number of studies that employing an active structural cooling system in combination with low density insulation requires less weight than with insulation alone. Active cooling is especially attractive for radiative shield design. Prior to selecting an active cooling system, factors such as refurbishment, development risk, simplicity in design, cost and the state-of-the-art requirements of the overall thermal protection system should be considered. An active cooling system is incorporated in TPS by specifying code SULAT = 20. Although

INSULATION WEIGHT (W/A) COMPARISON FIGURE 6.6-7



	qMAX	(THETA) θ
○	10	200
□	10	800
△	80	200
◊	80	800

a number of active structural systems have been proposed only the "porous strip channel" (or porous tube) and the BGRV water-gel blanket system are considered.

Insulation/Water Requirements - The insulation or water requirement is the same for both active cooling systems. (Gross factors are provided to account for increases in weight due to system inefficiency or contingency.) The simplifying logic served as the starting point for developing approximate relationships to estimate insulation and water weights. The temperature distribution was assumed linear, and the insulation homogeneous, from the surface to the wall.

A heat energy balance at the wall yields

$$\frac{K}{X}\left(T_{SURF} - T_W\right) = \dot{m}\Delta HVAP \qquad 6.6.4\text{-}9$$

since $W/A]_{ins} = \rho X$

and $W/A]_{liq} = \dot{m}\theta$

$$\frac{K\rho}{W/A]_{ins}}\left(T_{SURF} - T_W\right) = \frac{W/A]_{H_2O}}{\theta}\ \Delta HVAP \qquad 6.6.4\text{-}10$$

In past studies, we have found that the optimum combined weight of insulation and water occurs when

$W/A]_{ins} = W/A]_{H_2O}$ Solving for $W/A]_{ins}$ gives;

$$WINSW(I) = C\left\{SL\text{Ø}20 * \left[\frac{K e\theta\ (T_{SURF} - T_W)}{4.32 \times 107}\right]^{1/2} + C\text{Ø}N20\right\}^N \qquad 6.6.4\text{-}11$$

The values of C = 1.3 and N = 1.375 are used in TPS and were obtained by comparing this equation to computer predicted data as shown in Figure 6.6-8 assuming SLØ20 = 1.0 and CØN20 = 0. The water weight includes a 30% increase to account for load carrying structural heat shorts and is computed by;

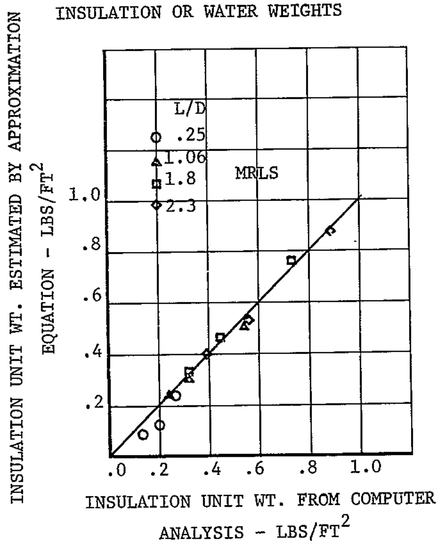
$$WH2\text{Ø}(I) = \left[H2\text{Ø}SL\text{Ø} * \frac{W}{A}\right]_{INS} * \left[1.3 + H2\text{Ø}C\text{Ø}N\right] * AREA(I) \qquad 6.6.4\text{-}12$$

The conductivity is evaluated at the average temperature and is;

$$K = SL\text{Ø}K20 * \left(\frac{T_{SURF} + T_w + 920}{2}\right) + C\text{Ø}NK20 \qquad 6.6.4\text{-}13$$

FIGURE 6.6-8

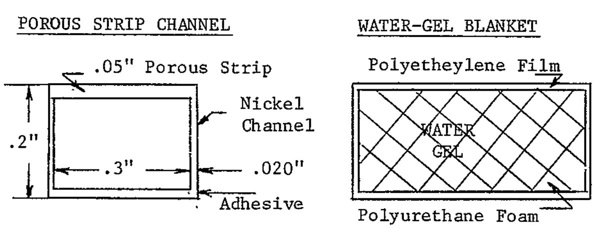
COMPARISON OF ESTIMATED AND PREDICTED
INSULATION OR WATER WEIGHTS



One factor contributing to the scatter of the data in Figure 6.6-8 is that the computer analysis assumes the conductivity to vary with ambient pressure as well as temperature. (In evaluating K for Equation 6.6.4-13, a mean pressure of 1. mm Hg, corresponding to an equivalent altitude of 150,000 feet, is recommended.)

Water Distribution and Supply Dry Weights - The porous strip channel and water blanket configurations are illustrated below:

FIGURE 6.6-9



The weight of channel and adhesive depends on the separation distance between tubes which in turn depends on the heating rate reaching the cooled wall. The separation distance "L" between tubes is given by

$$L = \left(\frac{2K_m\delta\,[T_{MAX} - T_t]}{\dot{m}\,\Delta H}\right)^{1/2} \qquad 6.6.4\text{-}14$$

where: (assume)

T_{MAX} = peak temp between channels ~ 275°F

T_t = temp at channel interface ~ 75°F

K_m = conductivity of cooled wall ~ $\frac{70 \text{ BTU-INCH}}{\text{Hr-Ft}^2\text{-°F}}$

δ = thickness of cooled wall ~ .035 inch

ΔH = heat of vaporization of H_2O ~ 1000 Btu/lb

$\dot{m}$ = water vaporization rate ~ lb/sec

L = separation distance between tubes ~ inch

Substituting the above values in Equation 6.6.4-14 yields an approximate relationship for determining the weight of porous strip channel.

$$WCHNL(I) = SCBNL * 15.4\sqrt{\dot{M}} * AREA(I) \qquad 6.6.4\text{-}15$$

where: SCHNL is an arbitrary constant (=0 for the water-gel blanket system).

The storage weight consists of either the water tank for the porous strip system or the dry fraction of the water-gel system, and is calculated by,

WTANK = TANKS * WH2ØT ** TANKE + TANKC   6.6.4-16

where:

TANKS ≐ .325   TANKE = .8   TANKC = 0.   for water tank

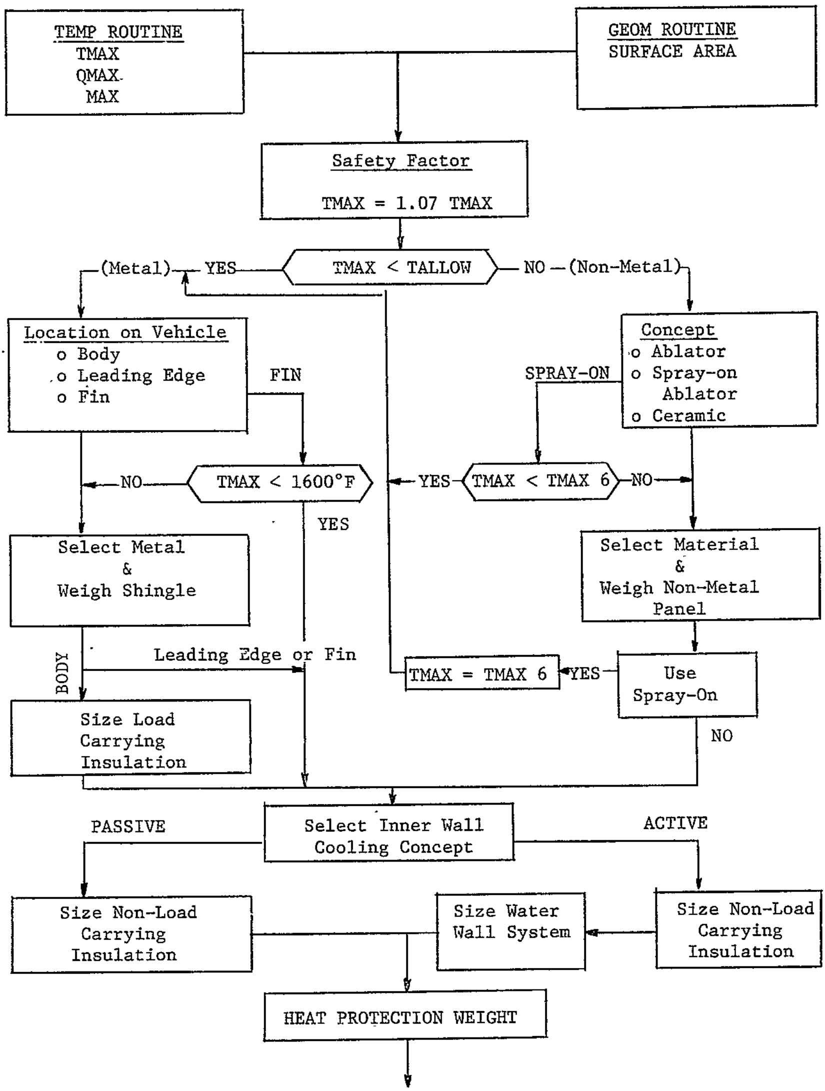
TANKS = .15   TANKE = 1.0   TANKC = 0.   for water-gel dry weight

6.6.5 Flow Diagram - The logic flow for this subroutine is shown in Figure 6.6-10.

6.7 Power (PWR) Subroutine - The POWER Subroutine estimates the weight of the auxiliary (non-propulsive) power systems in both the crew module and the mission module (or adapter) of either a ballistic or a lifting reentry vehicle. Th types of power systems included in the subprogram are battery (auto and/or manually activated), oxygen-hydrogen fuel cell, and monopropellant hydrazine auxiliary power unit. In addition, the weights of other types of systems can be used as direct inputs to the program. A weight estimation of the hydraulic system for the actuation of aerodynamic control surfaces of lifting type reentry configurations is also included.

TPS FLOW CHART

FIGURE 6.6-10



All electrical power requirements (except those for control surface actuation) are estimated external to the subroutine and are included in the inputs. Control surface actuation horsepower requirements are estimated by the subroutine.

In this subroutine, the control surfaces are actuated by dual hydraulic systems, each system capable of supplying the peak horsepower requirements. Each hydraulic system is powered by a completely separate power source, either battery or auxiliary power system.

This discussion covers limitations and selectors first. Then power requirements for movable surfaces, power generation system, and power conversion and distribution equations are discussed followed by an explanation of the derivation of some of the equations.

6.7.1 Program Limitations - As presently written, the capabilities of this subroutine are limited in the following three areas. If, and when, time permits, these limitations can be eliminated if the resulting extra flexibility seems necessary.

Hinge Moment Calculations - The equations for calculating the hinge moments of movable aerodynamic surfaces were written specifically for the NASA M2-F2 vehicle configuration and are not applicable for vehicles of other shapes.

Power Generation - The weight equations for the power generation equipment only covers batteries, fuel cells, and hydrazine fueled auxiliary power units. For other equipment, such as nuclear and solar dynamic or static devices, system weights must be estimated outside the loop and entered as a subroutine input.

There are no provisions for gaseous storage of fuel cell reactants as the program is presently written.

The auxiliary power unit can be sized to handle electrical loads as well as hydraulic loads but with the present equations, the APU is on only during reentry.

Power Conversion and Distribution - The only type of power conversion equipment considered in this subroutine is a hydraulic system using linear actuators for actuation of surface controls. There is no provision for sizing electric motors (except to drive the hydraulic pumps), power hinges, hydraulic motors, etc. Also, there is no provision for actuation of devices other than control surfaces and variable geometry wings. Also, there is no provision for changing the amount of redundancy in the hydraulic system. The equations are based on two identical, independent hydraulic systems, each capable of delivering the peak horsepower requirement.

6.7.2 Selectors - This program permits some flexibility in the selection of equipment used in the vehicle under consideration. This selection is accomplished in several ways. In some cases it is controlled by setting an input either equal to zero if there is no requirement for a particular type of equipment or to a finite number if the equipment is required. The selection between two types of similar equipment is done by changing the magnitude of an input (such as the specific weight). The third type of selection is by a computed selector. These are SAVE1, SAVE, and SAVE2.

SAVE1 = SELECT*TYPE
SAVE = SELECT - SAVE1
SAVE2 = SELECT/USEO

The value of (SELECT) determines whether or not there is an aero control actuation system and (TYPE) determines whether it is powered by an electric motor or by an auxiliary power unit.

Additional selectors, SLCT1, SLCT2, and SLCT3, are used to define the configuration of the electrical power system as shown in Table 6.7-1.

TABLE 6.7-1

Electrical Power System Configuration				Select Switch Position		
Ascent & Phasing Battery	Reentry Battery	Fuel Cell	Reactant Tanks	SLCT1	SLCT2	SLCT3
EV or MM	EV	-	-	0	0	0
-	-	EV	EV	1	0	0
-	EV	EV	MM	0	1	1
-	EV	MM	MM	0	1	0

6.7.3 Power Requirements for Actuation of Movable Surfaces - The equations for estimating power requirements for actuation of the movable surfaces provide for three sets of control surfaces (movable surfaces "A", "B", and "C") where each set can have single or multiple surfaces. Provisions for either a single or double pivot variable geometry wing are also included. These equations estimate the maximum hinge moment of each surface, and combine them with the maximum deflection rates to determine maximum horsepower per surface. These are combined to estimate peak and steady-state horsepower requirements for use in sizing batteries or an auxiliary power unit. The resulting peak and steady state horsepower requirements are those required to operate each of two redundant systems.

6.7.3.1 Variable Geometry Wing Actuation Power Requirement - To estimate the stall hinge moment for the variable geometry wing, (Equation 6.7.3-1) the drag during

```
HMVGWP = 33.75*(NVGP-1.0)*TR*B*B+(.000511*(NGVP-1.0)*LANDWT*LANDWT)/B+
         .009*B*B*WVGW                                              6.7.3-1
```

deployment is estimated and added to the inertia forces that result when extending the wing 90° in one second. The drag forces are based on a deployment q of 600 psf during equilibrium glide and with a profile drag coefficient of 0.2. The wing root thickness [TR], vehicle landing weight [LANDWT], wing span [B], and wing weight [WVGW] are inputs calculated in other subprograms. Since a one piece wing with a single pivot in the center has little if any resultant moment due to drag during deployment, the number of pivots in the wing [NVGP] is used to eliminate those portions of Equation 6.7.3-1 that are essentially zero in the single pivot wing. See Paragraph 6.7.6.1 for the derivation of this formula.

The horsepower required to rotate the variable geometry wing (Equation 6.7.3-2) is

```
HVGW = HMVGWP*VGDAV/31513.0                                         6.7.3-2
```

of the form (hinge moment)x(rotation rate) ÷ foot-lbs per second per horsepower. Since the rotation rate (VGDAV) is in degrees per second, the denominator contains the required conversion to radians per second.

6.7.3.2 Control Surface Actuators Power Requirements - The equations for estimating the stall hinge moments of the movable aerodynamic control surfaces are:

HMMSAS = .214*L**3 6.7.3-3

HMMSBS = .265*L**3 6.7.3-4

HMMSCS = .591*L**3 6.7.3-5

are based on the hinge moments for the M2-F2 vehicle defined in MAC Report E145, Minimum Manned Lifting Body Vehicle. The hinge moments are assumed to vary as the cube of the vehicle length.

The horsepower required to rotate movable surface "A", is;

HMSA = HMMSAS*FMSAMR/31513.0 6.7.3-6

is of the same form as Equation 6.7.3-2 described above. If there is not surface "A", the maximum deflection rate (FMSAMR) should be set to zero.

The hinge moment and horsepower equations for movable surface "B" and movable surface "C" are the same as for movable surface "A" except for variable names and are used in the same manner.

6.7.3.3 System Horsepower - System peak horsepower per system

SYSPH = .667*(HMSA+HMSB+HMSC) 6.7.3-7

is assumed to be 2/3 of the horsepower required to move surfaces "A", "B" and "C" simultaneously at the maximum hinge moments and rates. It is also assumed that this peak will provide adequate power for wing deployment. System steady state horse-power per system (Equation 6.7.3-8) is the horsepower required to maintain a leakage flow of

SSSH = .65626*(NMSA+NMSB+NMSC)+.1*SYSPH 6.7.3-8

.15 gallons per minute in both of the servo valves per actuator per hydraulic system plus 25% contingency on leakage flow horsepower in addition to 10% of the system peak horsepower. This allows either system to supply both sets of servo valves in the case of a failure of the other system.

6.7.4 Power Generation - This section discusses the equations used to estimate the weight of the power generation equipment in the spacecraft crew module and mission module or adapter. This subprogram is based on the following assumptions concerning the configuration of the power generation system:

1. The electrical loads, other than that required for aerodynamic control actuation, will be provided by either manually or atuomatically activated batteries, oxygen-hydrogen fuel cells, solar cells, or other type generation systems in both the crew module and the mission module.
2. If an aero control surface actuation system is required in the crew module it will be powered by either a battery or a hydrazine fueled auxiliary power unit.

The total electrical energy required by the main electrical bus (TEEMB) is calculated by equation 6.7.4-1

TEEMB = DCM*24.*(BEPFC + .085 *XNOMEN) 6.7.4-1

The basic electrical power requirement (BEPFC) input covers all electrical power other than that used to power aero control surfaces and that portion required by the ECS which is a function of the number of men. .085*XNOMEN accounts for that portion of ECS power that varies as the crew size varies. DCM*24 converts the power to energy.

6.7.4.1 Fuel Cell Weights - If fuel cells are to be used to supply the electrical power requirements in the spacecraft, the equations discussed in this section are used to estimate the cell weights and the reactant requirements. These equations assume the use of hydrogen oxygen fuel cells with cryogenic storage of the reactants.

Crew Module - The electrical output required (AEPFCC) from the fuel cell is calculated by equation 6.7.4-2 and is identical

AEPFCC = (BEPFC + .085*XNOMEN)*(SLCT1 + SLCT3) 6.7.4-2

to the power portion of equation 6.7.4-1 above. The selectors are discussed in Paragraph 6.7.2.

The dry weight and reactant weight for the crew module fuel cell system are given in equations 6.7.4-3 and 6.7.4-4 respectively.

WFCSCM = AEPFCC * SPWTC * RFFCC + 7. * TEEMB **.567*SLCT1 6.7.4-3

WFCRCM = 1.2*TEFCC*SLCT1 6.7.4-4

The crew module fuel cell net weight is given in equation 6.7.4-5

FCSWCM = WFCSCM + WFCRCM 6.7.4-5

The system dry weight includes the fuel cell battery, the reactant tankage, and controls and plumbing weight. The fuel cell battery weight is based on the average electrical output (AEPFCC), the battery specific weight (SPWTC), and a redundancy factor (RFFCC). The reactant tank weight

(5.96 *TEFCC ** .576) 6.7.4-6

is based on the graph of Figure 6.7-1 with the fluid weight based on the total energy required from the fuel cell (TEFCC), an oxygen to hydrogen weight ratio of 8 to 1, a specific reactant consumption of 1.0 lb per kilowatt-hour and a 20% contingency on reactant. Controls and plumbing are assumed to be a constant 20.0 lbs.

The weight of the reactants is based on the total energy requirements (TEFCC), a specific reactant consumption of 1.0 lb per kwh plus a contingency of 20%.

Mission Module - The weights of a fuel cell system in the mission module are given by equations which are the same as equations 6.7.4-2, 6.7.4-3 and 6.7.4-5 respectively for the crew module, except for the variable names, and are used in the same manner.

6.7.4.2 Battery Weights - Battery weights for electrical and electronic requirements and for squib requirements in both the crew module and the mission module and battery weights for the aerodynamic control system requirements in the crew module are estimated by the equations in this portion of the subprogram.

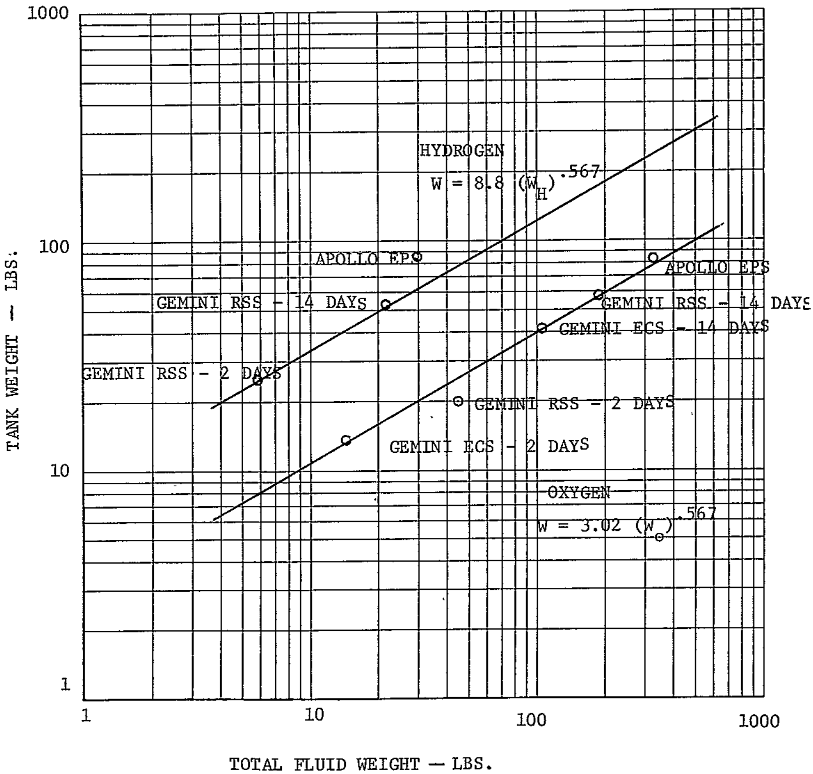
Crew Module - The weight for manually activated batteries (WMABMC) for the main bus loads, either the total loads if there are not fuel cells or for the peak loads and reentry loads if fuel cells are used, are calculated by equation 6.7.4-7.

WMABMC = TEEMB*FMABWC*RFMBBC*F*(1-SLCT1) 6.7.4-7

Battery weights are based on total energy required (TEEMB), battery specific weight (FMABWC), a redundancy factor (RFMBBC), a factor (F) which is used to divide this battery between the entry vehicle and the mission module.

FIGURE 6.7-1

REACTANT TANK WEIGHT



The weight of manually and automatically activated batteries for other crew module electrical loads except for squib and aero control surface actuation requirements are estimated by equations 6.7.4-8 and 6.7.4-9 respectively.

WMABSC = TEMABC * FMABWC * RFMABC   6.7.4-8

WAABSC = TEAABC * AABSWC * RFAABC   6.7.4-9

These equations are of the same general form as equation 6.7.4-7 above.

The equation to determine weight of the batteries to power both redundant aero control actuation systems (Equation 6.7.4-10)

WACPBC = SAVE1 * T * .733 * ACPBSW * RFACPB * (SYSPH + 3.0 * SSSH)   6.7.4-10

is of the same general form as above with the battery specific weight (ACPBSW) and the redundnacy factor (RFACPB) as the only inputs. The battery specific weight should take into account any inefficiency in the battery to determine the energy requirements. It is assumed that the control actuation system operates 25% of the time at peak horsepower and 75% of the time of steady state horsepower. It is also assumed that the efficiencies of the hydraulic pump and electric motor are 85% and 60% respectively. The system operating time [T] is the time required to descend from approximately 400,000 feet altitude. SAVE1 is a selector that sets the equation to zero if these batteries are not required and is generated elsewhere in this subroutine.

To include special power generation devices (such as isotope power supplies) that may be required in some vehicles, miscellaneous power generation system weights (Equation 6.7.4-11) can be generated similar to the approach used for batteries. Energy

WMPGSC = FMSTEC * FMSSWC * FMSRFC   6.7.4-11

requirements (FMSTEC), system specific weight (FMSSWC), and a redundancy factor (FMSRFC) are inputs to this equation.

The weight of the batteries required for squib initiation (WSBCM) is given by equation 6.7.4-12. For this study, this battery weight is assumed to be a constant

WSBCM = 20.0   6.7.4-12

20 lbs and is located in the entry vehicle. There are no squib batteries in the mission module.

Mission Module - The equation for the weights of manually activated batteries, automatically activated batteries, and miscellaneous power generation systems, are the same as their counterparts in the crew module, except for variable names, and are used in the same manner.

6.7.4.3 Auxiliary Power System - If an auxiliary power system is to be used to supply power for control surface activation, the equations discussed in this section are used to estimate the APU and fuel weights. These equations are based on the use of a hydrazine fueled system and are taken from curves in Sundstrand Proposal No. 2325A-P1 titled "Auxiliary Power Unit for Space Re-entry Vehicle", dated January 24, 1968.

The peak horsepower (Equation 6.7.4-13) and the steady state horsepower (Equation 6.7.4-14)

```
PHAPU = (SYSPH + PEPAPU) * SAVE                                   6.7.4-13
SSPAPU = (SSSH + SEPAPU) * SAVE                                   6.7.4-14
```

supplied by the APU are the sums of the hydraulic requirements and the electrical (PEPAPU and SEPAPU) requirements to be supplied. The weight of the APU (Equation 6.7.4-15),

```
WAPU = ((1.286 * PHAPU + 9.75) * 2 - (1.0 + .416 * SYSPH)
        * SELECT) * SAVE                                          6.7.4-15
```

based on Figure 6.7-2, includes gas generator, turbine, gear box, hydraulic pump, alternator, and power conditioning. Since the weight of the hydraulic pump is to be included in the weight of the aero control power system, the weight of the pump is subtracted from the APU weight. Equation 6.7.4-15 gives the weight for two APU's.

The specific fuel consumption for the unit when running at less than peak horsepower (Equations 6.7.4-16 and 6.7.4-17) is based on Figure 6.7-3. The total fuel requirement (Equation 6.7.4-18) assumes a duty cycle of 75% of the running

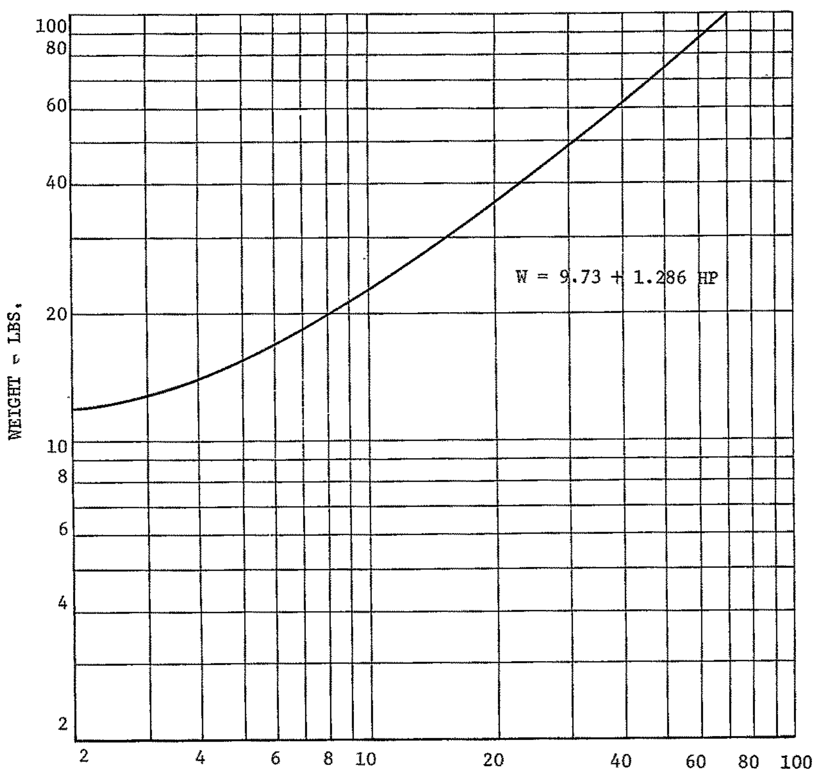
```
SSSFC = 4.2/((SSPAPU/PHAPU) ** .374) * SAVE                       6.7.4-16
IF (SSPAPU/PHAPU. GE. .33) SSSFC = 6.0/((SSPAPU/PHAPU) **.0524*SAVE
                                                                  6.7.4-17
```

time at steady state

```
WTFCM = (.25 * T * 6 * PHAPU + .75 * T * SSPAPU * SSSFC) *2*SAVE
                                                                  6.7.4-18
```

FIGURE 6.7-2

APU WEIGHT (LESS FUEL, TANKS & OIL RESERVOIR)



requirements and 25% at peak requirements. The fuel consumption at peak output is 6.0 pounds per horsepower-hour.

The dry weight of the fuel supply (Equations 6.7.4-19 and 6.7.4-20) is based on Figure 6.7-4 which assumes a small gas storage bottle for pressurizing

```
WDFS = (.1341*WTFCM + 6.28)* SAVE                          6.7.4-19
IF (WTFCM.GE.260.0) WDFS = (.081*WTFCM+33.9)* SAVE         6.7.4-20
```

the fuel tank for starting then using bleed gas from the decomposition chamber to maintain tank pressurization.

The total weight of the auxiliary power system is the summation of the component weights.

FIGURE 6.7-3

SPECIFIC FUEL CONSUMPTION

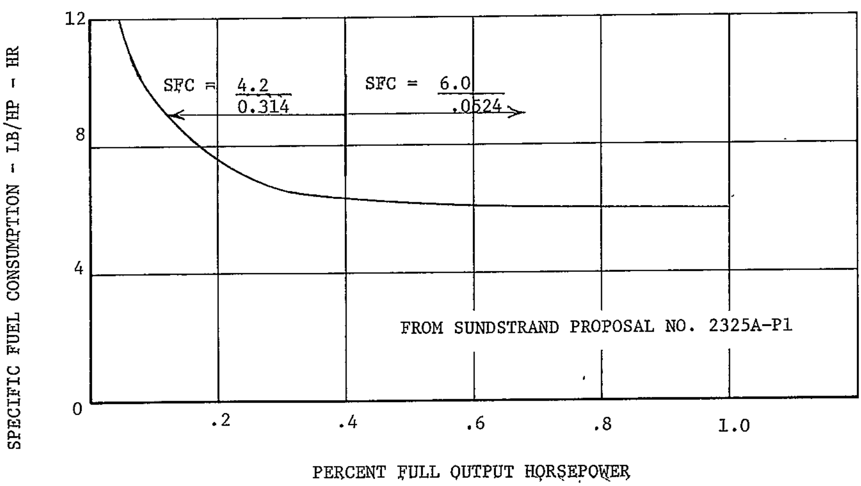
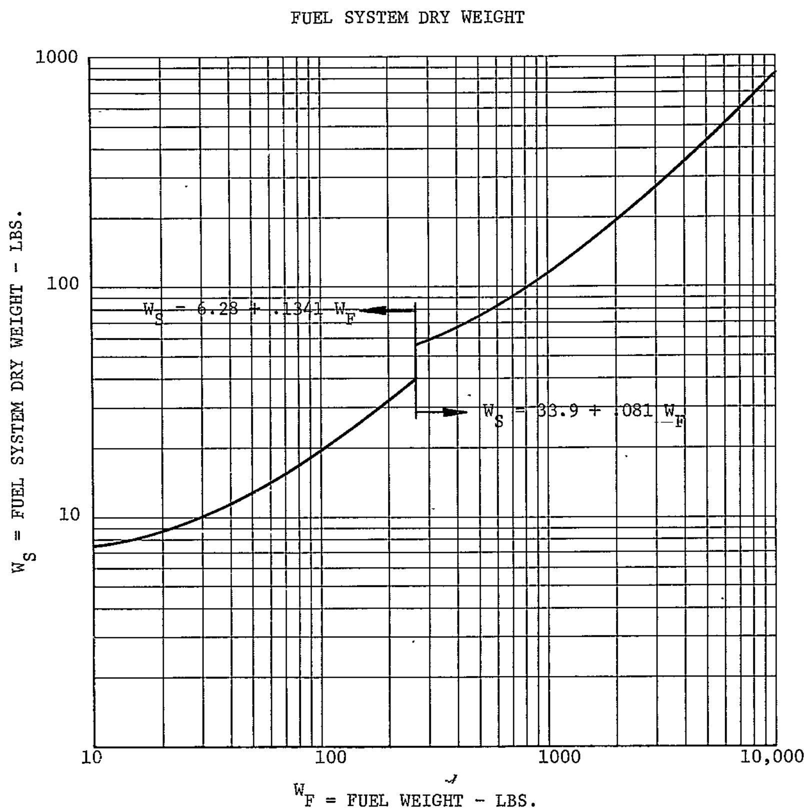


FIGURE 6.7-4


FUEL SYSTEM DRY WEIGHT
1000
100
10
10
100
1000
10,000
$W_S = 6.28 + .1341\ W_F$
$W_S = 33.9 + .081\ W_F$
W_S = FUEL SYSTEM DRY WEIGHT - LBS.
W_F = FUEL WEIGHT - LBS.

6.7.4.4 Power Generation System Wiring - The weight of the wiring for the power generation system (Equation 6.7.4-21) in the crew module is an empirical relationship based on the generation system equipment weight. Since the turbine

```
WPGSWC = .039*(WFCSCM+WMABSC+WAABSC+
         WSBCM+WSCSCM+WMPGSC)**1.298+.078*          6.7.4-21
         ((WAPU+WACPBC)/2)**1.298
```

and aero control battery weights are for two sets of equipment, those weights are divided by 2 and then the resultant wire weight is doubled. The equation for wire weight in the mission module is the same for the crew module except for the variable names.

6.7.4.5 Power Generation System Supporting Structure - The weight of the supporting structure for the power generation system (Equation 6.7.4-22) in the crew module is an empirical relationship based on the equipment weight.

```
WPGSSC = .875*(FCSWCM**.65+WMABSC**.65+
         WAABSC**.65+2(WACPBC/2)**.65+              6.7.4-22
         WSCSCM**.65+WMPGSC**.65+2*
         (WTAWCM/2**.65)
```

The equation for the support structure in the mission module is the same except that the variable names are different.

6.7.4.6 Power Generation System Total Weight - The total weight of the power generation system in the crew module (Eaution 6.7.4-23) is the summation of the weights of the various components.

```
TWPGCM = FCSWCM+WMABSC+WAABSC+WACPBS+
         WSCSCM+WMPGSC+WPGSWC+WPGSSC+               6.7.4-23
         WTAWCM+WSBCM+WMABMC
```

6.7.5 Power Conversion and Distribution - This section discusses the equations used to estimate the weight of the power conversion and distribution equipment in the spacecraft crew module and mission module or adapter and provides the substantiation for these equations.

6.7.5.1 Electrical Power Distribution System - The weight of the electrical power distribution system for the crew module and for the mission module are inputs to the subroutine.

6.7.5.2 Hydraulic System - The POWER subroutine is based on the use of dual redundant hydraulic systems to provide power for control surface actuation. Each system can provide the peak horsepower requirements of the vehicle. The hydraulic systems are powered by either a battery-electric motor combination or by a hydrazine fueled auxiliary power unit. Each of the following equations give the weight for two pieces or sets of equipment except for the control surface actuators.

The fluid volume of the actuators for the variable geometry wing actuation (Equation 6.7.5-1) is calculated from the angular rotation of the wing (VGWTRA),

AVDVGW = 24.*HMVGWP*VGWTRA*SAVE2 6.7.5-1

the hydraulic system maximum operating pressure (SMOP), and the maximum hinge moment. While operating pressure is an input, all weight equations are based on equipment designed for 3000 psi. The use of other pressures will induce some error in equipment weight. The fluid volume of the actuators for the movable surfaces is the same except for variable names. The total fluid volume of all actuators is given by Equation 6.7.5-2.

TVAF = SELECT*(AVDVGW+AVDMSA+AVDMSB+AVDMSC) 6.7.5-2

The weight of the hydraulic reservoirs (Equation 6.7.5-3) is based on Figure 6.7-5.

WTRES = SELECT*(6.2+.017*TVAF) 6.7.5-3

with the reservoir capacity equal to 10% of the fluid volume of all the actuators plus 10% of actuator volume for piston rod allowance and 25.7% for fluid expansion. Figure 6.7-5 is based on the boot-strap type reservoir.

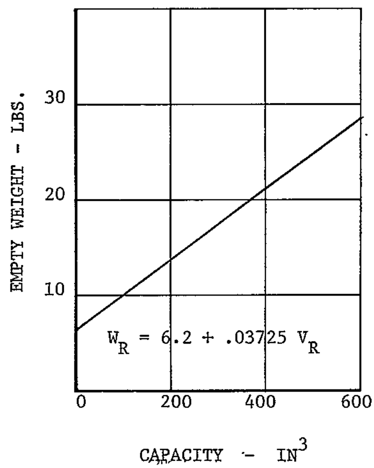
The hydraulic system contains filters in both the pressure and return lines and also in the pump case drain line. The equations for filter weight (Equation 6.7.5-4) assume that the pressure and return filters are the same size

WTSC = SELECT*(.001435*(ARG1+38.)**2.)*XN*2. 6.7.5-4

and that the largest filter used will handle the flow equivalent to 200 horsepower. Since the largest off-the-shelf pump has an output of approximately 200 horsepower, multiple pumps and multiple filters would probably be used in any system where the peak horsepower exceeds 200. Figure 6.7-6 shows the weight of 2 pair of filters

HYDRAULIC RESERVOIR WEIGHT

FIGURE 6.7-5



FILTER WEIGHT

FIGURE 6.7-6

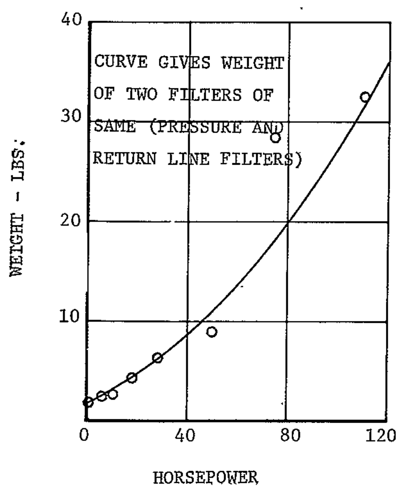
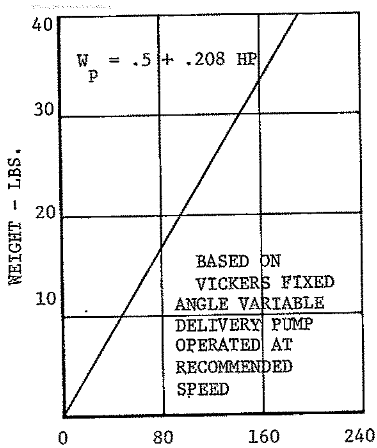


FIGURE 6.7-7

HYDRAULIC PUMP WEIGHT



The weight of the hydraulic fluid in the reservoirs is given in Equation 6.7.5-8.

WTFH = .457*WTAF*SELECT 6.7.5-8

The volume, and thus the weight, of the fluid in the reservoirs is assumed to be 45.7% of the volume of the actuators.

6.7.5.3. Hydraulic System Power Source - The hydraulic pump is powered either by an electric motor or an auxiliary power unit. The weight of the gas generator and turbine portion of the APU are calculated as part of the power generation system. The gear box weight, is calculated as part of the power conversion and distribution system.

For electric motor driven pumps, the weights of the motors are based on the curves of Figure 6.7-10. Equation 6.7.5-9 computes a motor weight based on steady state horsepower; Equation 6.7.5-10 on peak horsepower.

FIGURE 6.7-8

WEIGHT OF BARE ACTUATORS

(NO SERVO VALUES)

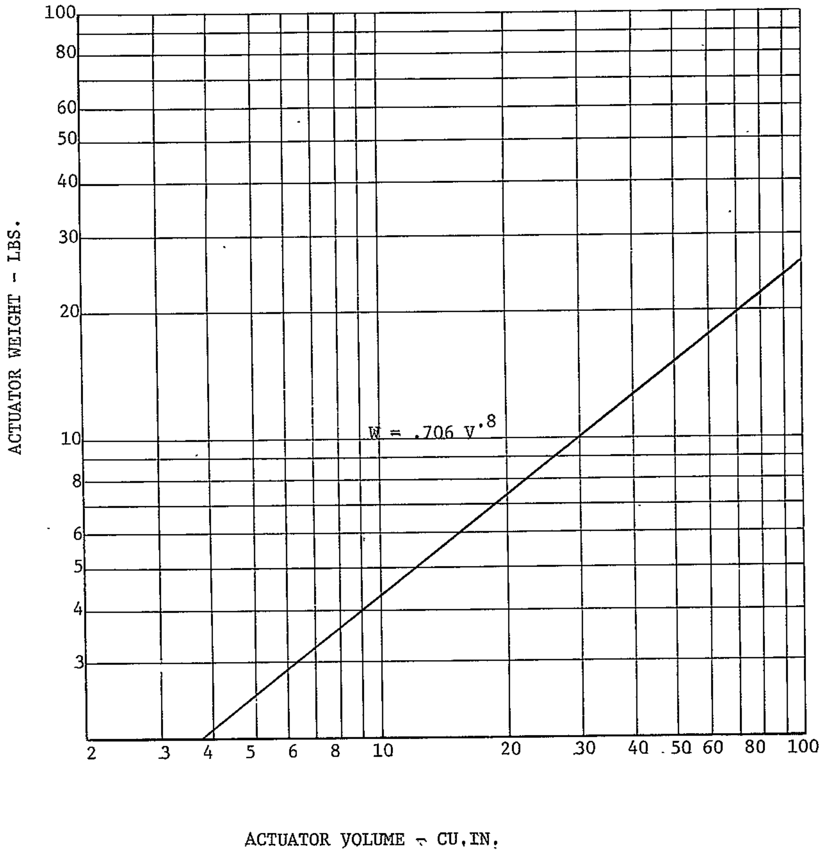
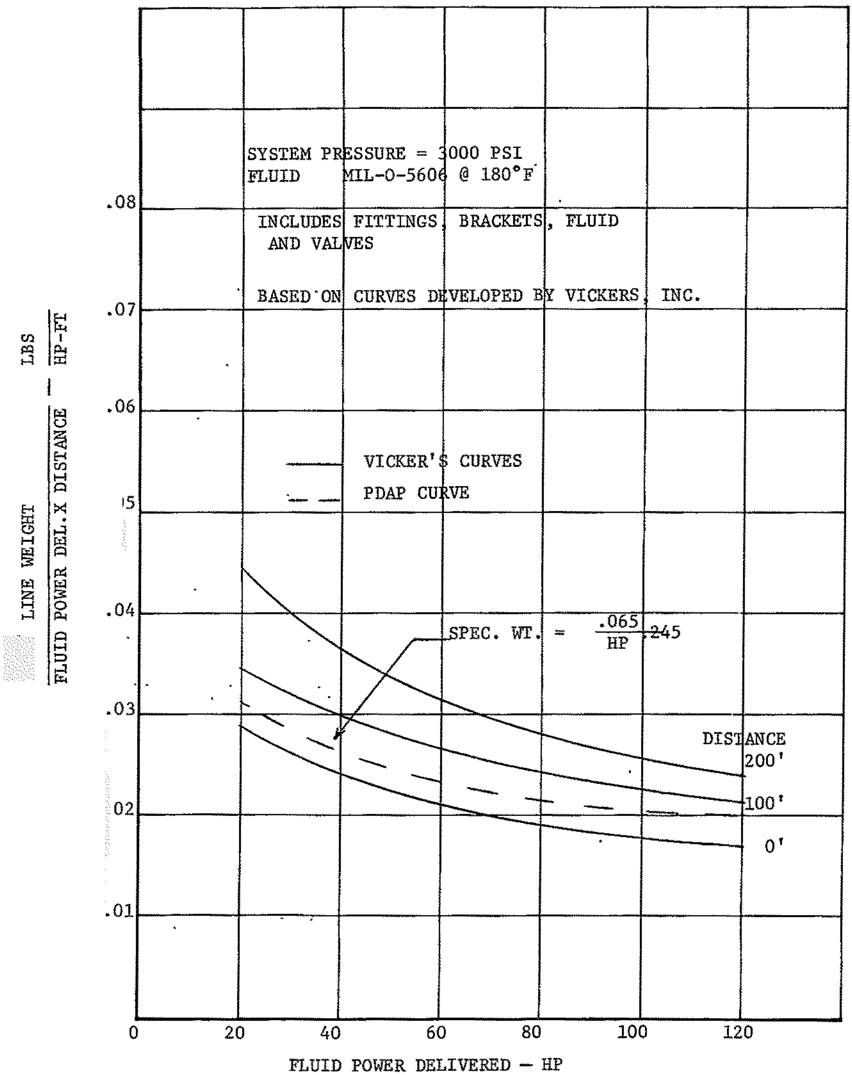


FIGURE 6.7-9

SPECIFIC WEIGHT OF HYDRAULIC POWER TRANSMISSION LINES



```
TEMP40 = 26.0*(SSSH/.85)**.587                      6.7.5-9
TEMPXX = 10.8*(SYSPH/.85)**.587                     6.7.5-10
IF(TEMP40.LT. TEMPXX) TEMP40 = TEMPXX               6.7.5-11
WTMOT = TEMP40*SAVE1                                6.7.5-12
```

the .85 factor in the denominator assumes the hydraulic pump is 85% efficient. Equations 6.7.5-11 and 6.7.5-12 choose the highest of the two computed weights as the motor weight.

The weight of the radio noise filters is based on the curves of Figure 6.7-10. Equations 6.7.5-13 and 6.7.5-14 compute weights based on steady state

```
TEMP41 = 5.0*(SSSH/.85)**.43                        6.7.5-13
TEMPXX = 2.1*(SYSPH/.85)**.43                       6.7.5-14
IF (TEMP41.LT.TEMPXX) TEMP41 = TEMPXX               6.7.5-15
WTRNF = TEMP41*SAVE1                                6.7.5-16
```

and peak horsepower requirements and Equations 6.7.5-15 and 6.7.5-16 choose the highest of the two.

6.7.5.4 Fly-by-Wire Wiring - The weight of wiring required to incorporate a fly-by-wire Flight Control System (6.7.5-17) assumes 1.0 lb of wire for each

```
WTWIRE = 2.18*L*SELECT                              6.7.5-17
```

foot of vehicle length (L)* plus 68% for connectors and 30% of wire and connector weight for installation. One pound per foot of vehicle length for wire weights assumes 100 two conductor, twisted, shielded wires with an average routed length equal to the vehicle length.

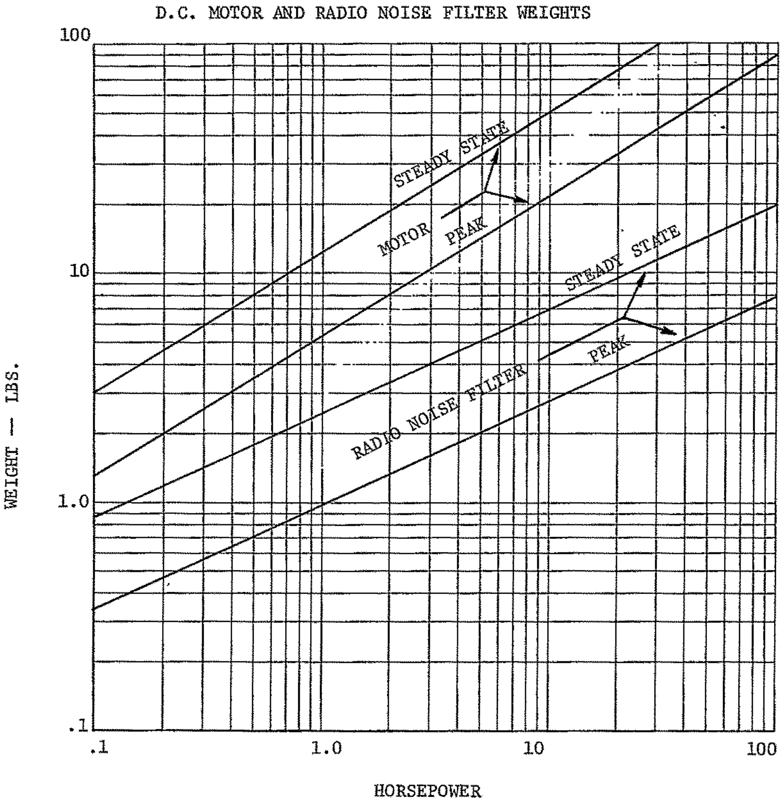
6.7.5.5 Power Distribution System Supporting Structure - The same empirical relationship used previously to estimate the weight of supporting structure is used for the power distribution system (Equation 6.7.5-18).

```
SSPDS = .875**SELECT*((WTRES+WTFH)**.65+WTSC**.65+WTPWRT**.65+
        (WTPUMP+WTMOT+WTRNF)**.65)                  6.7.5-18
```

6.7.5.6 Total Weight of Aero Control Power Distribution System - The weight of the power distribution system (6.7.5-19) is the summation of previously calculated weights.

```
WACPSC = WTRES+WTSC+WTFH+WTRWRT+WTWIRE+WTPUMP+WTMOT+WTRNF+SSPDS
                                                    6.7.5-19
```

FIGURE 6.7-10


D.C. MOTOR AND RADIO NOISE FILTER WEIGHTS
100
10
1.0
.1
WEIGHT — LBS.
.1
1.0
10
100
HORSEPOWER
STEADY STATE
MOTOR
PEAK
STEADY STATE
RADIO NOISE FILTER
PEAK

6.7.5.7 Total Weight of Power Conversion and Distribution - The total weight of the power conversion and distribution for the crew module Equation 6.7.5-20

$$WPCDCM = WEPSCO+WACPSC \qquad 6.7.5\text{-}20$$

and for the mission module (Equation 6.7.5-21) are summatiors of previously calculated weights.

$$WPCDMM = WEPSMO \qquad 6.7.5\text{-}21$$

6.7.5.8 Total Battery Weights - The total battery weights in the crew and mission modules are given by Equations 6.7.5-22 and 6.7.5-23.

$$WBSCM = WMABSC+WAABSC+WACPBC+WSBCM+WMABMC \qquad 6.7.5\text{-}22$$

$$WBSMM = WMABSM+WAABSM \qquad 6.7.5\text{-}23$$

6.7.6 Derivation of Equations - This section shows the derivations of some of the equations where the derivation is not readily apparent.

6.7.6.1 Stall Hinge Moment - Variable Geometry Wing (HMVGWR) - Two types of variable geometry wing are considered, a single pivot wing and a two pivot wing. For the two pivot wing the primary forces causing the hinge moment are due to profile drag and induced drag on the wing segments as they are deployed. Friction is not considered but is assumed to be accounted for by adding the moment resulting from the inertia force required to rotate the wing 90° in one second and by assuming that the entire wing is exposed i.e. outboard of the fuselage. On the single pivot wing the forces are more difficult to approximate but as the wing is deployed we would expect higher drag forces on the forward swept half of the wing than on the swept back half, with a resultant opening moment. Somewhat arbitrarily, however, it was elected to provide a hinge moment capable of rotating the wing 90 degrees in one second when only inertia forces are considered.

Symbols:

t_R = Root thickness - Ft

q = dynamic pressure at wing deployment = 600 psf

C_{D_o} = profile drag coefficient = .2

e = span efficiency factor = .9

LANDW = spacecraft landing weight - pounds

γ = sweep back angle of deployed wing

= 0° for single pivot wing

= 30° for two pivot wing

C_{D_i} = induced drag coefficient

A = frontal area of exposed semispan = Ft^2

S_w = Total theoretical wing planform area - Ft^2

B = Total wing span - Ft

AR = aspect ratio

HMVG = Stall hinge moment of variable geometry wing for each pivot - Ft-Lb = 1.5 x hinge moment

$$HMVG = (C_{D_o}\, q\, A \frac{B \cos \gamma}{2} + C_{D_i}\, q \frac{S_w}{2} \frac{B \cos \gamma}{2})\ 1.5$$

But: $$C_{D_i} = \frac{C_L^{\ 2}}{\pi e (AR)}$$

And for 1 g deployment:

$$C_L = \frac{LANDW}{q\ Sw}$$

Also: $$Ar = \frac{(B \cos \gamma)^2}{S_w}$$

And: $$A = (t_R)\ (B \cos \gamma)/2$$

Or: $$HMVG = 1.5\ (q \frac{B \cos \gamma}{2}) \left[C_{D_o} \frac{B\ (\cos \gamma)(t_R)}{2} + \frac{1}{2} \left(\frac{LANDW}{q\ Sw}\right)^2 \frac{(Sw)^2}{\pi x.9x(B\ \cos\gamma)^2}\right]$$

And for q = 600, cos γ = .866

$$MHVG = 1.5\ (22.5\ B^2 t_R + \frac{(LANDW)^2}{B} X \frac{1}{2940}) \qquad 6.7.6\text{-}1$$

Now consider the inertia forces only and rotate wing 90° in one second.

HMVG = 1 α

W(RADIAN/SEC) X 1.0 sec = 90/57.3 (RADIANS)

and assuming a linear acceleration:

$$\alpha = 2\ X\ 90/57.3\ rad/sec^2$$

$$I = mass\ X\ (radius)^2 = \frac{WVGW}{2g}\ \frac{.7B^2}{2}\ \frac{(WVGW)B}{16\ g}$$

$$HMVG = (WVGW) \frac{B^2}{16X32.2} \ X \ 2 \ X \ \frac{90}{57.3}$$

And again applying 1.5 factor to obtain stall hinge moment:

$$HMVG = .009 \ B^2 \ (WVGW) \qquad 6.7.6\text{-}2$$

Now Let NVGP = No. of variable geometry wing pivots and express the sum of equations 6.7.6-1 and 6.7.6-2 so that all terms except that due to inertia force disappear when NVGP = 1.

or: $$HMVG = 33.75 \ (NVGP-1.) \ B^2 (t_R) + .00051) \ X \ \frac{(NVGP-1)(LANDW)^2}{B} + .009 \ B^2 (WVGW)$$

6.7.6.2 Stall Hinge Moment - Movable Surface A (HMMSAS) - The stall hinge moment of the aero control system is estimated by calculating the force required at the center of pressure of the surface to rotate the vehicle at a given acceleration.

Symbols:

θ = Angular Acceleration

α = Angle between surface hinge line and vehicle rotation axis

X_a = Distance from vehicle nose to c.g. of surface

X_{cg} = Distance from vehicle nose to vehicle c.g.

I_p = Pitch moment of inertia

F_x = Force developed by the deflected surface

$X_{c.p.}$ = Distance from surface hinge line to c.p.

$$\theta = \frac{(X_a - X_{cg}) F_s}{I_p}$$

or

$$F_s = \frac{\theta \ I_p}{(Xa-Xcg)}$$

The resultant force is then

$$\frac{Fs}{\cos \alpha} = \frac{\theta \ I_p}{(Xa-Xcg) \cos \alpha}$$

and the hinge moment

$$H.M. = \frac{F_s X_{cp}}{\cos \alpha} = \frac{\theta I_p X_{cp}}{(Xa-Xcg) \cos \alpha}$$

Applying a buffet factor of 1.3 and a stall factor of 1.5

$$H.M. = \frac{1.95 \; \theta \; I_p X_{cp}}{(Xa-Xcg) \cos \alpha}$$

6.7.6.3 Actuator Weight - Figure 6.7-9 shows the weight of several simple actuators vs. their internal volume. For dual actuators, the volume is the sum of both cylinders. The volume of a tandem actuator can be approximated by

$$Vol. = \frac{2x12xHMMSASxFMSATA}{57.3xSMOPxNMSA}$$

where HMMSAS = Total stall hinge moment requirement for one set of control surfaces in foot-lbs

NMSA = Number of surfaces in set

FMSATA = Angle of rotation of surface in degrees

SMOP = Maximum system operating pressure in psi

It can be seen from Figure 6.7-9 that the weight of an actuator is approximately

$$Wt = .705(Vol.)^{.8}$$

6.7.7 Flow Diagram - The logic flow of this subroutine is shown in figure 6.7-11.

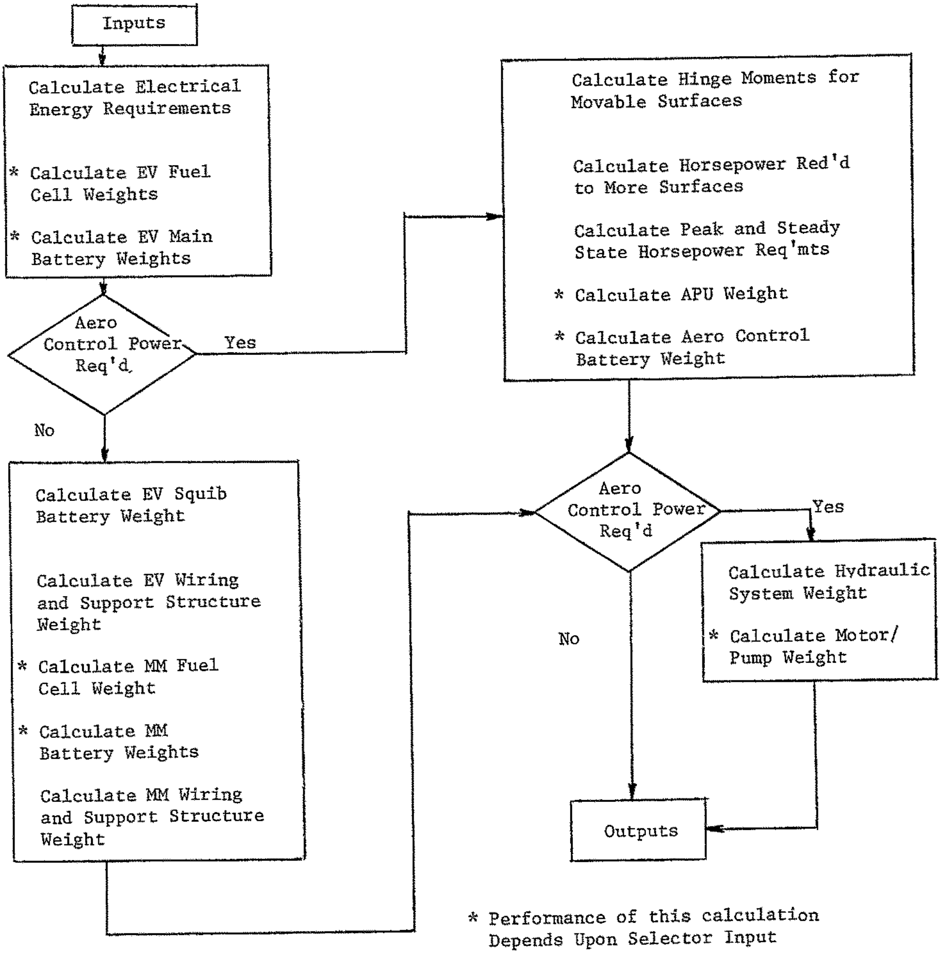
6.8 Environmental Control System Subroutine

SUMMARY - The ECS subroutine estimates the weight of equipment and expendables necessary to provide a habitable environment for the crew and thermal control of the equipment. For ease of analysis, the ECS is subdivided into four major subassemblies:

- o Atmospheric Gas Supply and Control
- o Gas Management and Processing
- o Heat Transport
- o Water Management

POWER SUBROUTINE FLOW CHART

FIGURE 6.7-11



Approximate weight relationships are used to estimate such items as:

1. oxygen and diluent requirements for breathing, leakage, pressurization and contingency
2. the tankage penalty for either high pressure or cryogenic storage
3. reconditioning of the oxygen stream with either LiOH or molecular sieve methods
4. thermal control of crew and equipment with space radiator and/or water boiler and
5. the water requirement and associated tankage.

In addition, gross methods for estimating the weight of pumps, heat exchangers, valves, mounting structure, coolant and circuitry are included to complete the ECS design.

In this program, the spacecraft system is comprised of a crew module (entry spacecraft) and a mission module which stores most of the expendables and equipment not needed during entry. The portion of ECS weight assigned to each module is controlled by input factors. The subroutine can size the ECS for either a mission or crew module alone or in combination. The approximate weight relationships developed herein are based primarily on empirical correlation of existing flight hardware data from Mercury, Gemini, Apollo, and LEM designs and from recent advance design studies.

In the following sections, ECS items are identified, and their approximate weight relationships are presented. The degree of data correlation obtained from analyses is shown.

6.8.1 ATMOSPHERIC GAS SUPPLY AND CONTROL ASSEMBLY - This assembly provides and controls life sustaining oxygen and inert diluent gas by manual and automatic controls. It replenishes the primary gas constitutents that are metabolically absorbed by personnel or lost through leakage; and thus, controls the partial and total environmental gas pressures for either a pure oxygen or dual gas atmosphere.

The following paragraphs present typical equations and supporting data to verify these equations.

6.8.1.1 OXYGEN AND DILUENT REQUIREMENTS AND APPORTIONMENTS - Oxygen is required for crew breathing and to maintain a pressurized environment. Diluent gas may be added to reduce the hazard of oxygen toxicity on long missions and/or reduce the hazard of fire.

Metabolic Oxygen - A requirement of 2 lbs per man-day of metabolic oxygen is used. This value is conservative and has been fairly well substantiated by past studies and actual missions.

The subroutine computes the total metabolic oxygen weight required, and apportions this quantity between storage form (cryogenic and high pressure) and between storage location (mission and crew module).

The total metabolic oxygen weight required is expressed by:

METOX = 2.0*ND

ND is the number of man-days in both the crew and mission modules.

Leakage Oxygen and Diluent - The subroutine computes separately the weight of oxygen and diluent required for leakage make-up, and apportions this quantity between storage form and between storage location.

The total leakage rate is an input variable. The following relationships yield the leakage of constituents in a dual gas atmosphere. Oxygen leakage in the crew module is given by the following typical equation:

LKOXCM = LKORCM *POX/POXR * DCM*(M/M + MDIL)

Similarly, the diluent leakage is

LKDLCM = LKORCM * PDIL/POXR*DCM*(MDIL/(MDIL + M)

where:

LKOXCM = Oxygen leakage from crew module, pounds
LKDLCM = Diluent leakage from crew module, pounds
LKORCM = Total leakage rate from crew module, pounds per day
POXR = Total pressure, psia
POX = Cabin oxygen partial pressure, psi
PDIL = Cabin diluent partial pressure, psi
MDIL = Diluent molecular weight, lb/mol
M = Oxygen molecular weight, lb/mol
DCM = Duration of men in crew module, days

Repressurization Oxygen and Diluent - The weight of gas required for pressurization is related to the pressured volume, gas partial pressure, and number of pressurizations.

The total cabin pressurized volume is obtained from the geometry subroutine. With this volume and with the number of repressurizations as inputs, the weight of oxygen and diluent required is determined. These quantities are apportioned between storage form (cryogenic and high pressure) and between storage location (mission and crew module).

The weight of oxygen is given by:

PRSOX = M*POX*VOL*REFIL/(R*T)

= .005625*POX*VOL*REFIL

Similarly, the weight of diluent is:

PRSDIL = MDIL*PDIL*VOL*REFIL/(R*T)

= .0001758*MDIL*PDIL*VOL*REFIL

where:

PRSOX and PRSDIL = Weight of oxygen and diluent for pressurization, pounds.

M and MDIL = Molecular weights of oxygen (32) and the diluent, pounds/mol.

POX and PDIL = Partial pressures of oxygen and diluent, psi

VOL = Total pressurized cabin volume, cubic feet

REFIL = Number of refills

R = Universal gas constant, 1545 lb-ft/°R-mol

T = Absolute temperature (assumed, 530°R)

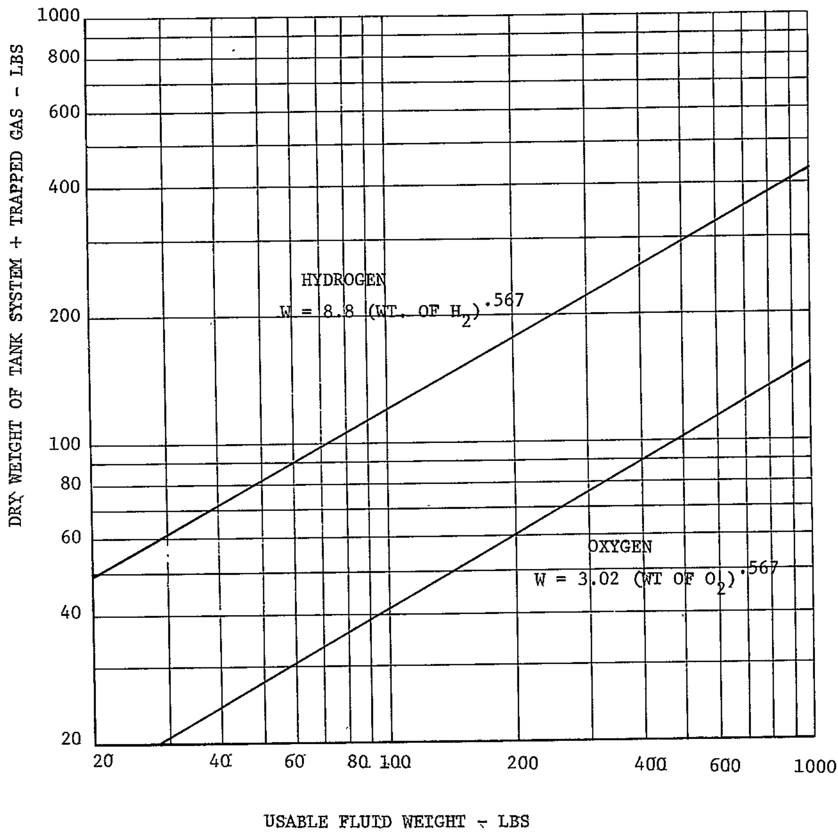
Trapped Gas and Contingency - In normal tank operation a fraction of the stored gas will not be recoverable. The amount of unrecoverable gas depends on the final gas pressure and temperature.

The subroutine computes trapped gas weight only for high pressure oxygen and diluent storage, and contingency gas weight for both high pressure and cryogenic gas supplies. Trapped gas weight in cryogenic supply tanks is not computed since this quantity is included in the tank weight as shown in Figure 6.8-1.

Secondary Oxygen - The subroutine computes secondary oxygen requirements. A secondary oxygen source is needed to supply oxygen for the crew during reentry and/or emergency conditions at a high flow rate.

FIGURE 6.8-1

CRYOGENIC TANKAGE WEIGHT SUPERCRITICAL STORAGE



This supply is always stored in the crew module as a high pressure gas. If flow rate and entry time are specified, the weight of oxygen is simply:

SECOX = FLOW*ENTIME*NCM

where:

FLOW is the flow rate, pounds per minute per man

ENTIME is the entry time, minutes

NCM is the number of crew in the reentry or crew module

6.8.1.2 TANKAGE REQUIREMENTS - Oxygen or diluent gas will be stored in either cryogenic or high pressure form. The fraction of fluid stored in either form is assigned by input factors. It is presumed that all of the secondary oxygen is stored as a high pressure gas, in order to facilitate the delivery of oxygen during entry.

The subroutine computes the following:

a) Volume of high pressure storage of oxygen and diluent located in the crew and mission modules.
b) Tank weight for high pressure and cryogenic oxygen and diluent storage located in crew and mission modules.
c) Summation of high pressure and cryogenic tank weights located in the crew and mission modules.

High Pressure Tankage - The high pressure tankage weight relationship in pounds is

TNKOKW = DENS/STRES*TNKSF*TNKOXV

where:

TNKSF = A parameter which includes the tank shape and design pressure design pressure (TNKSF = $P_t\lambda$), psi

P_t = Shell operating pressure, psi

λ = Shape factor, dimensionless

TNKOXV = Tank useable volume, ft^3

STRES = Allowable stress of tank metal at operating pressure, psi

DENS = Density of tank material, lb/ft^3

Assuming the high pressure gas is filled at 100°F, the useable gas volume is found from the ideal gas law to be:

TNKOXV = GASOXM*R*T/(M*TPOX) = 188. *GASOXM/TPOX where:

GASOXM = Weight of oxygen, pounds

R = Universal gas constant, 1545 ft lb/°R

T = Absolute temperature, assumes 560°R

TPOX = Tank fill pressure, psia

Cryogenic Tankage - The cryogenic oxygen and hydrogen equations shown below are based on the hardware correlation shown in Figure 6.8-1. Note that the correlation is based on useable cryogenic weight and assigns the trapped fluid to the tank dry system weight.

TNKOXC = 3.02 *(wt. of O_2)**.567 for Oxygen

Tank Weight = 8.8 *(wt. of H_2) **.567 for Hydrogen

The tankage weight relationships for nitrogen and helium were derived by noting that for equal useable volume the weight of storable fluid is proportional to the respective boiling point (or fill) densities, i.e.,

$$\text{Volume} = \frac{\text{Wt. of } O_2}{\text{Density of } O_2} = \frac{\text{Wt. of } N_2}{\text{Density of } N_2}$$

and

$$\frac{\text{Wt. of } H_2}{\text{Density of } H_2} = \frac{\text{Wt. of He}}{\text{Density of He}}$$

Substituting tank weight equations yields the tankage equations for nitrogen and helium.

Tank Weight = 3.68 *(wt. of N_2) **.567 for Nitrogen

Tank Weight = 6.35 *(wt. of He) **.567 for Helium

6.8.1.3 VALVES, CIRCUITRY, AND MOUNTING STRUCTURE - Although these items are often labeled "miscellaneous items" they constitute a sizeable portion of the ECS weight. The weight estimation techniques are based entirely on correlation of data from previous flight systems and advance design studies.

The subroutine computes the mounting weights for the sum of all tanks located in the crew and mission modules.

Valves - The weight of valves associated with the Gas Supply and Control Assembly located in the Crew Module (VLGSCM) is computed from the following:

VLGSCM = (VALVS/2) - VLGSMM

where:

VALVS = Total weight of valves for the Gas Supply and Control Assembly and the Gas Management and Processing Assembly, pounds

VLGSMM = Weight of valves associated with the Gas Supply and Control Assembly located in the MM, pounds

The factor of 1/2 is applied because the valve weight for the Gas Supply and Control Assembly and for the Gas Management and Processing Assembly is assumed to be equal.

The following equation for the total valve weight of both the Gas Supply and Control Assembly and the Gas Management and Processing Assembly (VALVS) is based upon correlation of Mercury, Gemini and Apollo ECS data and modified by point design studies.

where: VALVS = .77 *(29. + 12.*N) pounds

N = Number of men in CM and MM.

Circuitry, Lines and Fittings - The relationship used to estimate the weight of circuitry, lines and fittings associated with the Gas Supply and Control Assembly located in the Crew Module (CLFGSC) is:

CLFGSC = GSMCMT * CLF/WECST

where: GSMCMT = Total Gas Supply and Control Assembly weight in Crew Module without weight of circuitry, lines and fittings.

CLF/WECST = Ratio of circuitry, lines and fittings weight for the total ECS to the total weight of the ECS

The ratio is obtained from the following equations which are based on the correlation data shown in Figure 6.8-2.

For total ECS weight (WECST) less than 2390 lb:

$$\frac{CLF}{WECST} = .20 - 6.28 *10E\text{-}6*WECST$$

For ECS weight greater than 2390 lb:

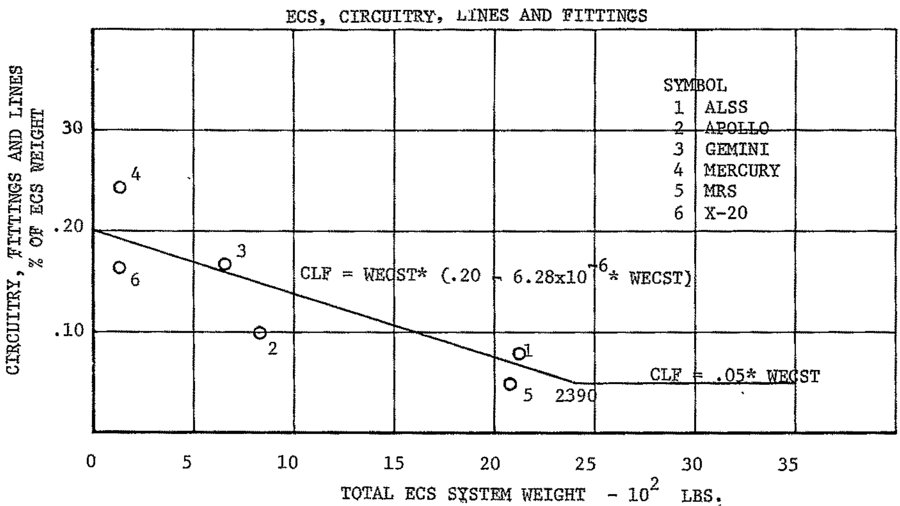
$$\frac{CLF}{WECST} = .05$$

where: WECST = total weight of ECS system without the circuitry lines and fitting weight, pounds.

CLF = Circuitry, lines, and fitting weight for total ECS, pounds

Mounting Structure Weight (W_{MS}) - The following empirical relationship for estimating mounting weight was derived from the correlation of Gemini data shown in Figure 6.8-3.

FIGURE 6.8-2



$$W_{MS} = 0.875*W_{Eq}**.65$$

where:

W_{MS} = mounting structure weight, pounds

W_{Eq} = weight of equipment to be mounted, pounds

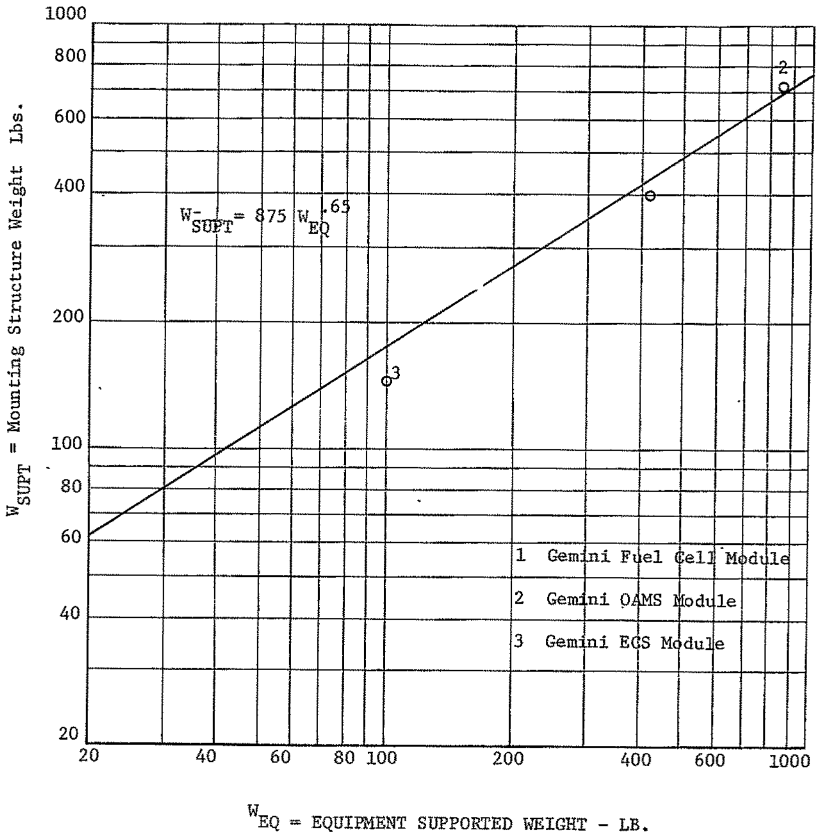
6.8.1.4 TOTAL WEIGHTS - The subroutine computes the total weights in each of crew and mission modules of tankage (cryogenic and high pressure), of gas supplies (cryogenic and high pressure), of valves, of circuitry, lines and fittings, and of mounting provision. A contingency weight is also included in the total system weight.

6.8.2 GAS MANAGEMENT AND PROCESSING ASSEMBLY - This assembly provides for atmospheric gas constituent control, suited and unsuited personnel cooling, and gas circulation.

The equations used to compute Gas Management and Processing Assembly weights are based upon theory and actual hardware experience from Mercury, Gemini and other programs and studies.

Figure 6.8-3

MOUNTING STRUCTURE WEIGHT



6.8.2.1 CARBON DIOXIDE AND ODOR REMOVAL SYSTEM (WCO2) - Two options of carbon dioxide and odor removal systems are available to the user: a lithium hydroxide system and a regenerable molecular sieve system.

The subroutine computes the weight of a lithium hydroxide and/or a regenerable molecular sieve system and apportions the system weight between use locations (crew and mission modules).

Data from Gemini, Mercury and from Reference (4) was used to derive the weight equation given below.

WCO2 = K*(3.7*ND)+(1-K)*(14.3*NCM + .233*ND+86.)

where:

WCO2 = CO_2 removal system weight, pounds

K is the selection factor

K = 1, LiOH System

K = 0, Molecular Sieve System

6.8.2.2 PERSONNEL COOLING REQUIREMENTS - Cooling is required for the crew in the cabin environment and in the closed suit environment. It is assumed that cooling will be provided by a suit heat exchanger and, optionally by a liquid cooled garment.

Heat Exchanger and Water Separator (HEX) - The heat exchanger and water separator cools the gas circulated in the suit loop or, for open suit operation, in the cabin loop and removes condensed water vapor.

The subroutine computes the weight of the heat exchanger and water separator assembly and apportions the assembly weight between use locations (crew and mission modules).

The hardware correlation shown in Figure 6.8-4 is used for this estimate.

HEX = 11.2*N**.77

where:

HEX = Heat exchanger and water separator weight, pounds

N = Number of men

Liquid Cooled Garment (LCG) - A cooling garment may be worn by the crew members. Coolant circulates through tubes woven in the garment. A data correlation based on Hamilton Standard's quoted weight for a one man system and a nine man point design is used to estimate the LCG weight (see Figure 6.8-5).

The subroutine computes the total weight of liquid cooled garments and apportions the weight between use locations (crew and mission modules).

The equation for estimating the weight of liquid cooled garments is:

LCG = M*(3.5*N+6.5)

FIGURE 6.8-4

HEAT EXCHANGER AND WATER SEPARATOR

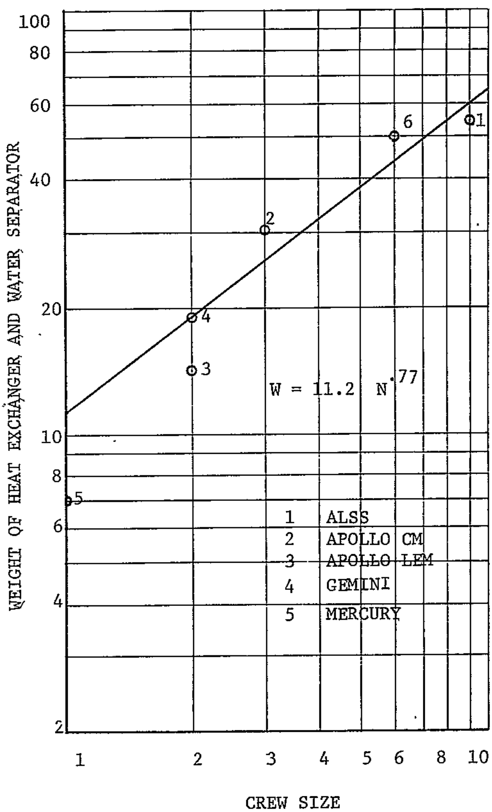
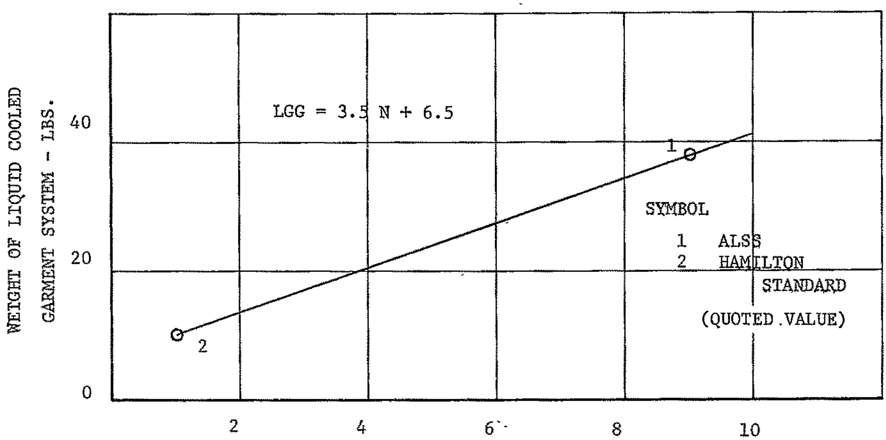


FIGURE 6.8-5

LIQUID COOLED GARMENT



NUMBER OF MEN

where:

LCG = Liquid cooled garment weight, pounds

N = Number of men

M = 0 if liquid cooled garment is not used

M = 1 if liquid cooled garment is used

6.8.2.3 ATMOSPHERIC GAS CIRCULATION REQUIREMENTS - Compressors, valves, circuitry, lines and fittings are required for gas circulation.

Suit Compressor and Converter (STCMP) - A compressor provides the pressure differential needed to circulate gas in either the suit loop or, for open suit operation, in the cabin loop. A converter changes D.C. current to A.C.

An alternate standby compressor and converter is provided in parallel circuit with the operating compressor and converter for redundancy.

The subroutine computes the weight of dual compressors and conveters and apportions the weight between use locations (crew and mission modules).

A correlation of hardware and point designs, shown in Figure 6.8-6 provides the weight estimation equation for dual compressors and converters as follows:

STCMP = 3.6*N+11.7

where:

STCMP = weight of dual suit compressors and converters, pounds

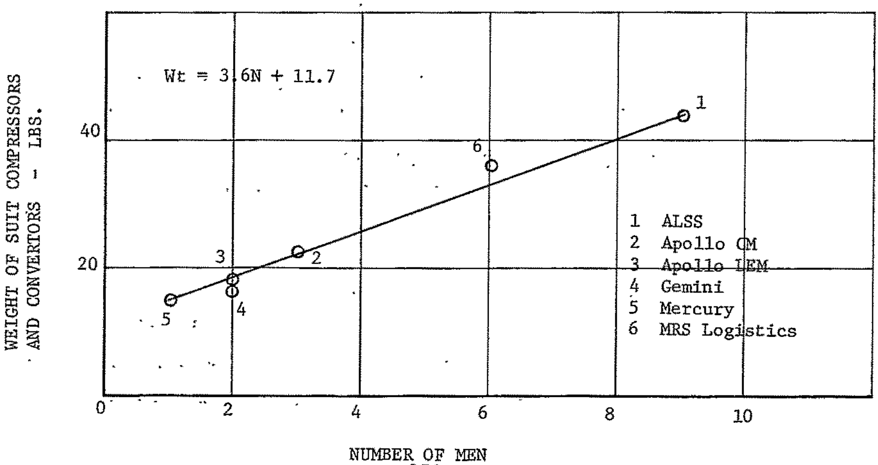
6.8.2.4 VALVES, CIRCUITRY AND MOUNTING STRUCTURE - Estimating equations for these items were discussed in Section 6.8.1.3.

6.8.2.5 TOTAL WEIGHTS - The subroutine computes the total weights in the crew and mission modules of the carbon dioxide removal system (lithium dioxide or regnerable molecular sieve), of dual suit compressors and converters, of heat exchanger and water separator assembly, of liquid cooled garments, and of valves, circuitry and mounting structure. A contingency weight is also included in the total system weight.

6.8.3 HEAT TRANSPORT LOOP - The heat transport subassembly provides a means of removing and dissipating heat from equipment and men, and providing controlled temperatures. Heat removal may be accomplished by radiation to space, or by evaporative cooling.

Figure 6.8-6

DUAL SUIT COMPRESSOR AND CONVERTER WT.



The equations used to compute the heat transport assembly weights are empirically correlated but are based on theoretical relationships. Typical equations, and supporting data, are given in paragraphs 6.8.3.1 through 6.8.3.3.

6.8.3.1 HEAT TRANSPORT EQUIPMENT - The subroutine first computes the heat loads. The electrical power dissipation is input from Subroutine POWER and it is assumed that all the equipment power is dissipated as waste heat. In addition, the batteries and fuel cells are assumed to have efficiency of 90% and 67% respectively, and the resulting waste heat is included in the heat transport assembly load. The crew metabolic load was assumed to be a constant value of 620 Btu/hr per man, which includes 120 Btu/hr per man heat of CO_2 absorbtion. These loads then provide a basis for sizing the heat transport equipment.

Pump Package Assembly (PUMP) - The pump package assembly consists of the pump, power supply and accumulator. The following equation was derived by correlating data from studies and the Gemini (A and B) Flight Systems. The degree of correlation is shown in Figure 6.8-7.

$$PUMP = .56 \ (Q \ MAX)^{.25} + 24.$$

where:

Pump = Weight of the pump, power supply and accumulator, lb.

Q MAX = Sum of equipment and metabolic peak heating rates, Btu/hr.

Coldplates (CLDP) - Most heat generating electrical equipment is mounted on coldplates in order to maintain the equipment within acceptable temperature limits. A correlation of Gemini individual coldplates data, shown in Figure 6.8-8 provided the following weight relationship:

$$CLDP = 1.5 \ As + .15$$

The surface area (As) is related to the maximum equipment heat transfer rate by:

$$As = \frac{\dot{Q} \ ELC}{h_{eff} \Delta T}$$

where:

CLDP = Coldplate weight, lb

As = Coldplate heat transfer area, ft^2

$\dot{Q}$ ELC = Equipment peak heating rate, Btu/hr

$h_{eff}\Delta T$ = Effective heat flux rate per unit area, Btu/ft^2-hr

Combining these equations and assuming a nominal value of 500 Btu/ft^2-hr for

FIGURE 6,8-7

PUMP PACKAGE CORRELATION

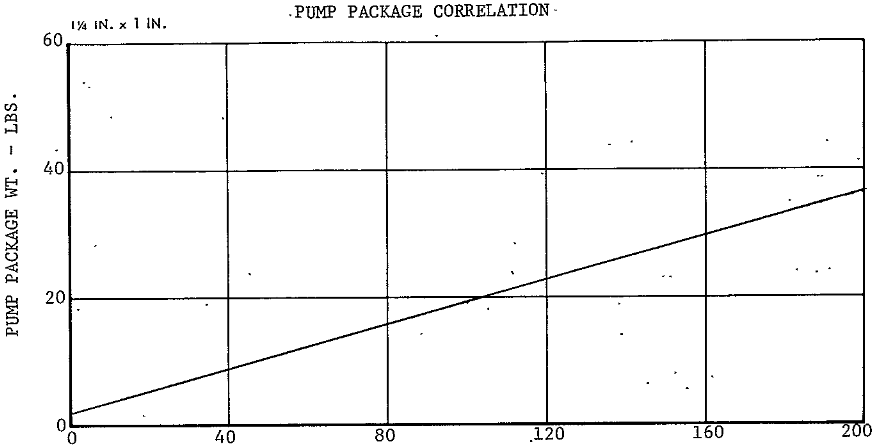
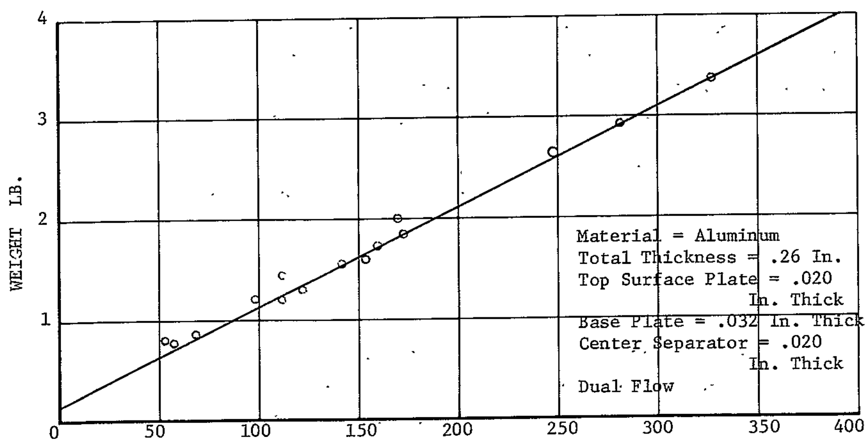


Figure 6.8-8

GEMINI COLD PLATE DATA



$h_{eff}\Delta T$ and "tuning" the resultant equation with study data resulted in:

CLDP = .003*Q ELC + .15

Cabin Heat Exchanger (HX) - The cabin heat exchanger is a small weight item. Since no simple correlation was found, a constant weight of 6.0 lbs is presumed regardless of system size:

HX = 6.0

6.8.3.2 HEAT DISSIPATION EQUIPMENT - The heat absorbed in the heat transport circuit is dissipated either by a water boiler or radiator or both. If a radiator is selected it will dissipate all the heat provided enough radiator area is available. If not, then a water boiler dissipates the portion the radiator cannot. If a radiator is not selected than all the heat is dissipated by water boilers. It should be noted that a radiator can be allowed only if the vehicle includes a mission module.

Radiator - The radiator weight is determined with an emitted heat flux of 45 Btu/hr ft^2, a value based on the nominal Gemini radiator performance. The effectiveness of the radiator is given by:

QRAD = 45*SA

where:

QRAD = Radiated heat flux, Btu/hr

SA = Available radiator surface area, ft^2 (From subroutine GEOM)

The value of QRAD is compared with the system heat load. If it is less than the load then a water boiler is included in the system. If QRAD is greater than the load then the radiator area is calculated with the relation:

RA = QAVE/45.

where:

RA = The actual radiator surface area, ft^2

QAVE = Average heat load plus 5%, Btu/hr

The weight of radiator structure (WRSTR) is calculated with:

WRSTR = .3*RA*(1.-NRSTR) + 1.1*RA*NRSTR

where:

NRSTR = 0 specifies the Gemini integrated radiator structure

NRSTR = 1.0 specifies the MOL nonload carrying radiator structure

Water Boilers - The water boiler consist of an internal heat exchanger and pressure regulator. Consideration of the Gemini and Apollo water boilers led to the following empirical equations. It is assumed that the CM boiler is used only for re-entry and the MM boiler is used for peak heating

The weight is given by:

BOILCM = .0026 *(QELCCM + QMETCM) + 10.

where:

BOILCM = CM boiler weight, lb

QELCCM = CM equipment waste heat, Btu/hr

QMETCM = CM metabolic waste heat, Btu/hr

BOILMM = .0026 *FRDIS*QMAXMM + 10

where:

BOILMM = MM boiler weight, lb

FRDIS = fraction of peak heat load dissipated by water boilers, dimensionless

QMAXMM = peak total system heat load, Btu/hr

6.8.3.3 MISCELLANEOUS - Methods for calculating the weight of mounting structure and circuitry are discussed in Section 2.3. The valve weight for the heat transport assembly was assumed to be 10% of the heat transport equipment weight.

6.8.4 WATER MANAGEMENT ASSEMBLY - This system includes the drinking water, the water vaporized in the water boiler, and the tankage and plumbing necessary for storage and fluid transfer. An input factor assigns the fraction of weight stored in the crew and mission modules.

6.8.4.1 DRINKING WATER REQUIREMENTS - A requirement of 6.2 lbs of water per man-day was assumed, based on Gemini experience. In addition, a 10% safety factor was assigned to account for contingency and "trapped" water in the tank. The requirements are given by:

DRIKCM = 6.2*NCM*DCM*1.1

DRIKMM = 6.2 *NMM*DMM*1.1

where:

DRIKCM = Drinking water required in CM lb

NCM = no. crewmen in CM

DCM = days crew in CM

DRIKMM = Drinking water required in MM, lb

NMM = no. crewmen in MM

DMM = day crew in MM

6.8.4.2 BOILER WATER REQUIREMENTS - The boiler water required is calculated based on the fraction of total energy dissipated by boiling, and a latent heat of vaporization of 1000 Btu/lb. The requirements are calculated with the relations:

H2OCM = .001*QDISMM

H2OMM = .001*QDISCM

where:

H2OCM = CM boiler water required, lb

QDISCM = heat dissipated in CM by boiler, Btu

H2OMM = MM boiler water required, lb

QDISMM - heat dissipated in MM by boiler, Btu

6.8.4.3 FUEL CELL WATER PRODUCTION - The amount of water produced by operation of the fuel cells is credited to the water management assembly. These quantities are called from Subroutine POWER and included in the computation of total water requirements.

6.8.4.4 WATER TANK WEIGHTS - A hardware correlation of tank weights from the Mercury, Gemini, and Apollo programs, shown in Figure 6.8- 9 is used to estimate stretchable bladder for water-gas separation.

WATK = .96*(DRNK + H2O) **.56

where:

WATK = water tankage weight, lb

DRNK and H2O = total water requirements, lb

6.8.4.5 TANKAGE EQUIPMENT REQUIREMENTS - Tankage equipment required includes valves, circuitry, lines, fittings and mounting structure. The circuitry and mounting structure weights are based on the method of Section 6.8.1.3. The valve weight is assumed to be 20% of the tankage weight.

6.8.5 AERODYNAMIC CONTROL SYSTEM COOLING ASSEMBLY - This assembly provides cooling of the hydraulic system used to move the aerodynamic control surfaces. The assembly consists of a water boiler heat exchanger and a water supply tank and associated equipment.

6.8.5.1 WATER BOILER - The water boiler weight was calculated from the relation:

ACBOIL = .0026 ACPWH + 10

where:

ACBOIL = boiler weight, lb

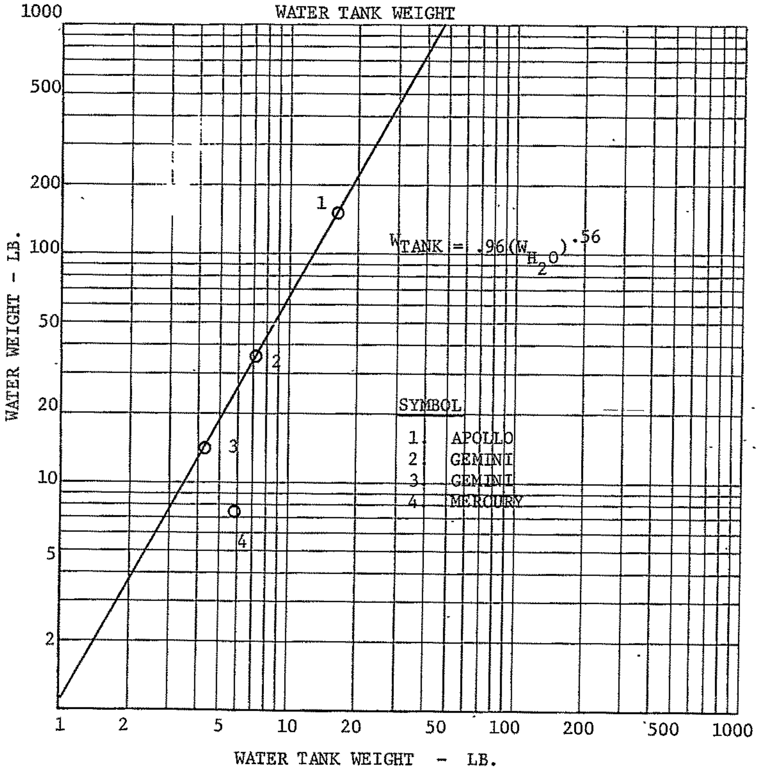
ACPWH = waste power, Btu/hr (calculated from Subroutine Power data)

6.8.5.2 COOLING WATER REQUIREMENT - The water requirements are calculated from:

H2OAC = .001* ACPWE

where:

FIGURE 6.8-9


WATER TANK WEIGHT
1000
500
200
100
50
20
10
5
2
WATER WEIGHT - LB.
1
2
5
10
20
50
100
200
500
1000
WATER TANK WEIGHT - LB.
$W_{TANK} = .96(W_{H_2O})^{.56}$
SYMBOL
1. APOLLO
2. GEMINI
3. GEMINI
4. MERCURY
1
2
3
4

H2OAC = cooling water weight, lb

ACPWE = waste energy, Btu (calculated from Subroutine AERO data)

6.8.5.3 WATER TANK - The water tank weights are based on the empirical relation developed from Figure 6.8-10.

WATKAC = .95*H2OAC**.56

where:

WATKAC = weight of aero control cooling water storage tank, lb.

H2OAC = weight of aero control cooling water, lb

6.8.5.4 MISCELLANEOUS - The method of calculating the weight of mounting structure, and circuitry, lines and fittings is discussed in Section 6.8.1.3. The valve weight is assumed to be negligible and is not computed.

6.8.6 FLOW DIAGRAM - The logic flow for this subroutine is shown in Figure 6.8-10.

6.9 CARGO SUBROUTINE - This subroutine determines cargo weight, its distribution between reentry module and mission module, and the associated mounting for the cargo. Cargo weight can be either an independent or dependent variable. The option of setting cargo as a dependent variable is achieved. by making DCARGO (1) a 2.0 (1.0 indicates independence) and DCARGO (2) equal to the booster effective launch weight capability (rather than total cargo weight in the independent case). If the requirements on the vehicle are such that a negative cargo would be required to meet the input effective launch weight, the cargo weight is set equal to zero and allowed to converge to an answer in excess of the effective launch weight capability.

After the total cargo weight is determined, it is distributed to the Reentry Module and Mission Module at launch based upon input decimal fractions. Cargo weight at reentry is calculated for both Mission Module and Reentry Module as a decimal times the total cargo weight at launch plus (or minus) a fixed input weight. Cargo, as spoken of above, is truely useful cargo. In addition to this weight, support weight is calculated for both Reentry Module and Mission Module cargo using the equation:

$$\text{Wt. Supt.} = .875\ (\text{Cargo Wt.})^{.65}$$

The larger of either the cargo weight at launch or entry is used in the above equation. The sum of cargo weight plus supports for Reentry Modules and Mission Module at Launch and Entry is the output which the Mass Property Subroutine requires. The flow diagram is shown in Figure 6.9-1.

ENVIRONMENTAL CONTROL SYSTEM

ECS INPUT DATA

ATMOSPHERIC GAS SUPPLY

Oxygen & Diluent Requirements
- o Metabolic O_2
- o Leakage
- o Repressurization
- o Contingency
- o Secondary O_2

Storage Place
- % Crew Module
- % Mission Module

Storage Form
- % CRYO., % High Pressure

Cryogenic Tank Weight
- O_2, N_2, He, H_2

High Pressure Tank Weight

Mounting Wt. Circuitry

Valves

GAS MANAGM & PROCESS

CO_2 Removal System
- o LIOH
- o Regenerative

Equipment Weights
- o Suit Compressor & Power
- o Suit Heat Exchang.
- o LCG
- o Valves & Sensors
- o Mounting Struct.
- o Circuitry

Storage Place
- % Crew Module
- % Mission Module

HEAT TRANSPORT LOOP

Peak & Total Metabolic & Equipment Heat Loads

CM Heat TR. Equip:
- o Pump Pack, Accum.
- o Cold Plates, HX
- o Water Boiler
- o Valves, Mount, Struct
- o Circuitry
- o Coolant

MM Heat Trans Equipment
- o Pump Pack, Accum.
- o Cold Plates, HX
- o Radiator &/or H_2O Boiler Heat Dissipation
- o Valves, Mount.Struct.
- o Circuitry
- o Coolant

WATER MANAGEMENT

Water Requirements
- o Drinking
- o Water Boiler

Dry System Requirements
- o Tankage
- o Mounting Structure
- o Circuitry
- o Valves

Storage Place
- % Crew Module
- % Mission Module

AERO CONTROL COOLING

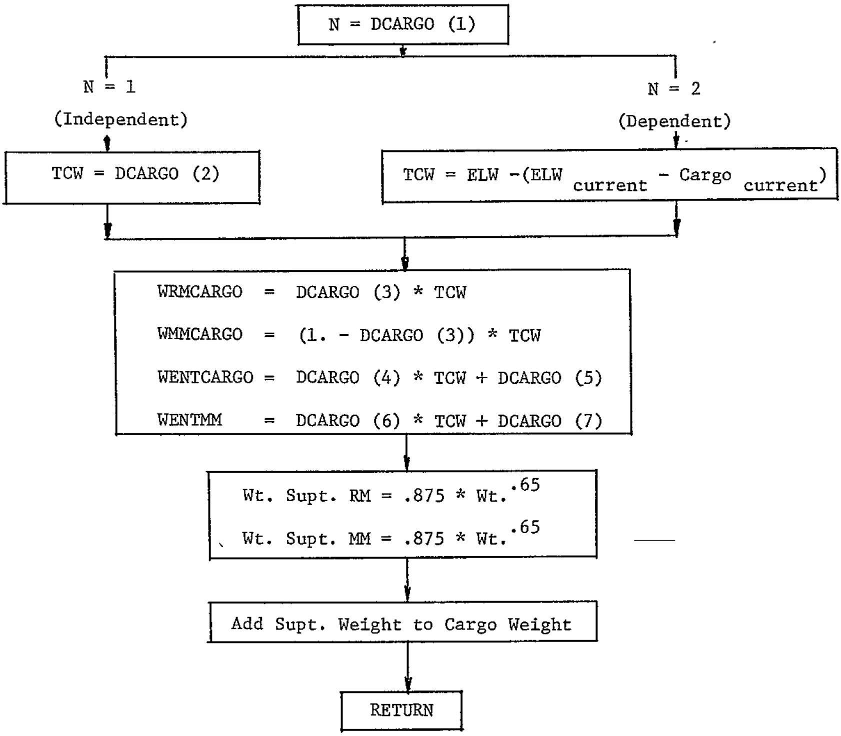
- o Boiler Weight
- o Water Required
- o Tankage
- o Circuitry

TOTAL ECS WEIGHTS

FIGURE 6.8-10

FIGURE 6.9-1

CARGO FLOW DIAGRAM



6.10 Miscellaneous Subroutine - This subroutine computes the weight of items which are not assignable to any other subroutine. This routine adds mounting and wire weights to component weights which have to be inputted. Weights are computed for these systems:

Docking

Recovery Aids

Ordnance and Separation Provisions

Range Safety

Guidance and Navigation

Instrumentation

Communication

Crew Station Controls

Flotation

The weight for each system is calculated for the crew and mission module separately.

6.10.1 Equation Discussion - The following equations are used to compute the various weights of the miscellaneous subsystems not computed elsewhere.

```
1.   WDOCK = DM(1)*DOCKW**DM(2)+DM(3)
2.   WDOCKA = DM(4)*DOCKW**DM(5)+DM(6)
3.   WREC = DM(7)*LANDW**DM(8)+DM(9)
4.   WFLOAT = DM(10)*LANDW**DM(11)+DM(12)
5.   WORD = 9.*(DM(13)+DM(14))+2.*DM(15)+(DM(16)+Pi*DSCA)*DM(17)
5.(A)  WORD = WORD+31.
6.   WORDA = 9.*DM(18)+2.*DM(19)+DM(20)*(DM(21)+Pi*DAB*DM(18))
7.   WRSFTY = DM(22)
8.   WGN = DM(23)+.412*DM(23)**.65+.039*DM(23)**1.298
9.   WGNA = DM(24)+.412*DM(24)**.65+.039*DM(24)**1.298
10.  WIN = DM(25)+.412*DM(25)**.65+.039*DM(25)**1.298
11.  WINA = DM(26)+.412*DM(26)**.65+.039*DM(26)**1.298
12.  WCOM = DM(27)+.412*DM(27)**.65+.039*DM(27)**1.298
13.  WCOMA = DM(28)+.412*DM(28)**.65+.039*DM(28)**1.298
14.  WCSC = DM(29)+.412*DM(29)**.65+.039*DM(29)**1.298
15.  WCSCA = DM(30)+.412*DM(30)**.65+.039*DM(30)**1.298
16.  WRSFTA = DM(31)
```

The equations define the weights for the nine subsystems noted in 6.10.

6.10.1.1 Equation 1 defines the weight of the docking system in the crew module.

Eq. 1: WDOCK = DM(1)*DOCKW**DM(2)+DM(3)

WDOCK = M(1) = DOCKING SYSTEM WEIGHT [LBS.]

DM(1) = COEFFICIENT FOR DOCKING SYSTEM WEIGHT:

.9 FOR DOCKING RINGS, 1.7 FOR DOCKING FORKS

DOCKW = MP(61,4) = VEHICLE WEIGHT AT DOCKING [LBS.]

DM(2) = EXPONENT FOR DOCKING SYSTEM WEIGHT; NORMALLY = .5

DM(3) = CONSTANT FOR DOCKING SYSTEM WEIGHT FOR FIXED WEIGHT SYSTEM.

A diagram of the docking system which is the basis for this equation is shown in Figure 6.10-1. The values of various DM(3) constants is shown in Figure 6.10-2.

6.10.1.2 Equation 2 is analogous to Eq. 1 except this one is used for the mission module.

Eq. 2: WDOCKA = DM(4)*DOCKW**DM(5)+DM(6)

6.10.1.3 Equation 3 defines the weight of the recovery system in the crew module.

Eq. 3: WREC = DM(7)*LANDW**DM(8)+DM(9)

WREC = M(3) = RECOVERY SYSTEM WEIGHT [LBS.]

DM(7) = COEFFICIENT FOR RECOVERY SYSTEM WEIGHT

LANDW = MP(61,7) = VEHICLE WEIGHT AT LANDING [LBS.]

DM(8) = EXPONENT FOR RECOVERY SYSTEM WEIGHT.

DM(9) = CONSTANT FOR RECOVERY SYSTEM WEIGHT.

6.10.1.4 Equation 4 defines the weight of the floatation system in the crew module.

Eq. 4: WFLOAT = DM(10)*LANDW**DM(11)+DM(12)

WFLOAT = M(4) = FLOATATION SYSTEM WEIGHT [LBS.]

DM(10) = COEFFICIENT FOR FLOATATION SYSTEM

LANDW = MP(61,7) = VEHICLE WEIGHT AT LANDING

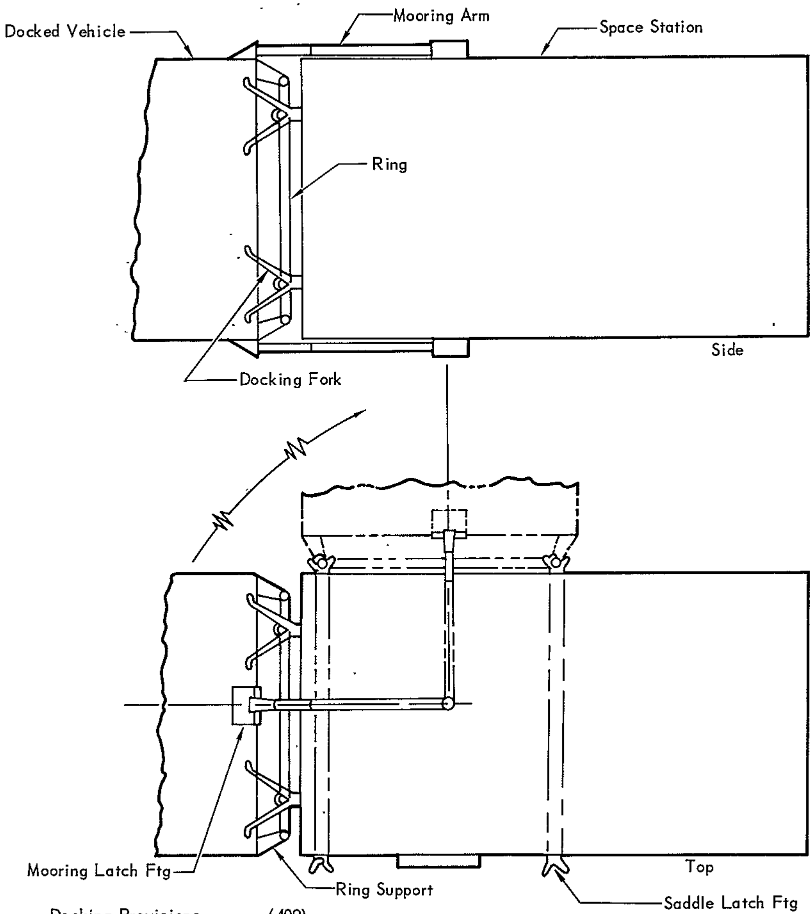
DM(11) = EXPONENT FOR FLOATATION SYSTEM

DM(12) = CONSTANT FOR FLOATATION SYSTEM

6.10.1.5 Equation 5 defines the weight of the ordnance and separation provisions. Nine pounds are allowed for batteries for each separation. Two pounds are allowed for charge for each jettisoned item. The weight of the shape charge necessary to separate the crew and mission modules is based on the coefficient

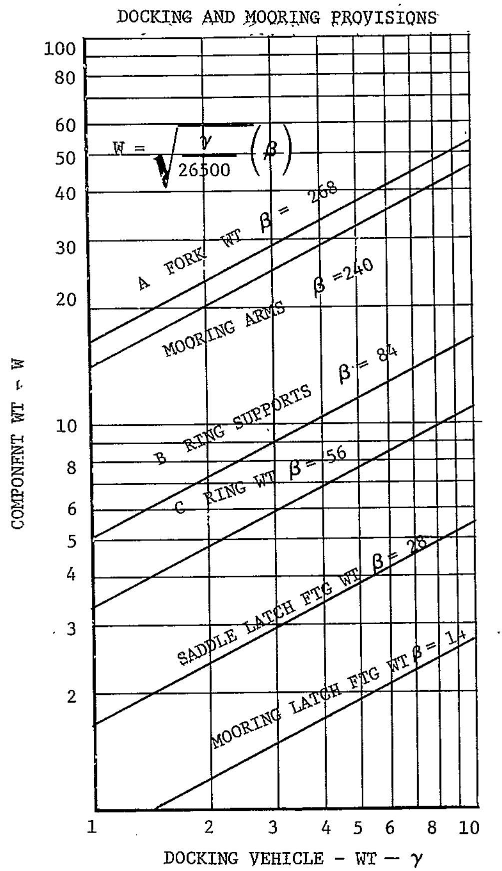
MOORING AND DOCKING PROVISIONS

FIGURE 6.10-1



Docking Provisions	(409)
Ring Install	141
Ring	56
Ring Supports	85
Forks	(268)
Mooring Provisions	(282)
Mooring Arms	240
Mooring Latch Ftg	14
Saddle Latch Ftg	28

FIGURE 6.10-2


DOCKING AND MOORING PROVISIONS
$W = \sqrt{\frac{\gamma}{26500}}\,(\beta)$
A FORK WT β = 268
MOORING ARMS β = 240
B RING SUPPORTS β = 84
C RING WT β = 56
SADDLE LATCH FTG WT β = 28
MOORING LATCH FTG WT β = 14
COMPONENT WT - W
DOCKING VEHICLE - WT — γ
100
80
60
50
40
30
20
10
8
6
5
4
3
2
1
2
3
4
5
6
8
10

for the type of shape charge [DM(17)], and the perimeter of the sections to be separated.

Eq. 5: WORD = 9.*(DM(13)+DM(14))+2.*DM(15)+(DM(16)+π*DSCA)*DM(17)

WORD = M(5) = ORDNANCE + SEPARATION PROVISIONS WEIGHT [LBS.]

DM(13) = NUMBER OF SEPARATIONS ON CREW MODULE

DM(14) = NUMBER OF SEPARATIONS ON M.M. USING C.M. POWER

DM(15) = NUMBER OF JETTISONED ITEMS ON CREW MODULE

DM(16) = PERIMETER OF BLOW OFF DOORS [FT.]

DSCA = DIAMETER AT C.M.-M.M. INTERFACE [FT.]

DM(17) = COEFFICIENT FOR TYPE OF SHAPE CHARGE [LB/FT.]

When the ordnance and separation/systems are used (Eq. 5 >0.), then 31. lbs are added to account for items not included in Eq. 5, such as attachment devices, circuitry, etc.

Eq. 5.A.: WORD = WORD+31.

6.10.1.6 Equation 6 is analogous to Eq. 5. It defines the weight of the ordnance and separation provisions in the mission module.

Eq. 6: WORDA = 9.*DM(18)+2.*.DM(19)+DM(20)*(DM(21)+π*DAB*DM(18))

WORDA = M(6) = ORDNANCE AND SEPARATION PROVISIONS WEIGHT IN THE M.M. [LBS.]

DM(18) = NUMBER OF SEPARATIONS ON M.M.

DM(19) = NUMBER OF JETTISONED ITEMS ON M.M.

DM(20) = COEFFICIENT OF TYPE OF CHARGE [LB./FT.]

DM(21) = PERIMETER OF M.M. DOORS [FT.]

DAB = G(80) = DIAMETER OF BOOSTER CYLINDER [FT.]

6.10.1.7 Equation 7 defines the weight of the range safety equipment, for the crew module. This value is an input variable.

Eq. 7: WRSFTY = DM(22)

6.10.1.8 The following equations define the weights of component systems. The weight for each system is based on the general formula:

SYSTEM WT. = COMP. WT. + .412 (COMP. WT.)$^{.65}$ + .039 (COMP. WT.)$^{1.298}$

Where .412 (COMP. WT.)$^{.65}$ is the weight of the mounting provisions and .039 (COMP. WT.)$^{1.298}$ is the wire weight. This equation shown in Figure 6.10-3. Figure 6.10-4 shows the wire weight equation.

Equation 8 defines the weight of the guidance and navigation system in the C.M.

FIGURE 6.10-3

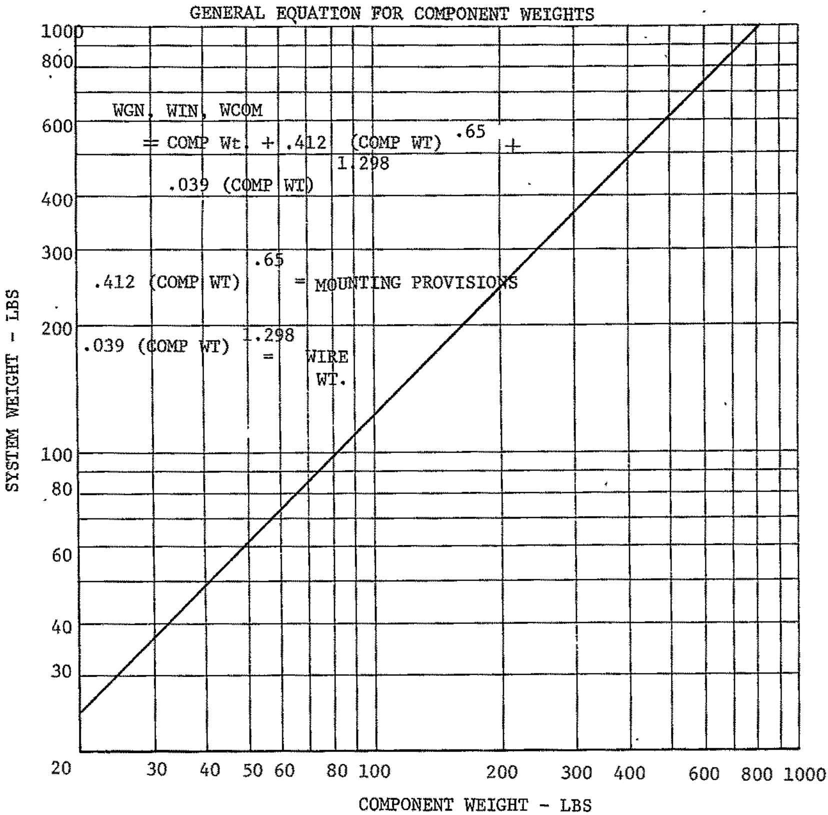
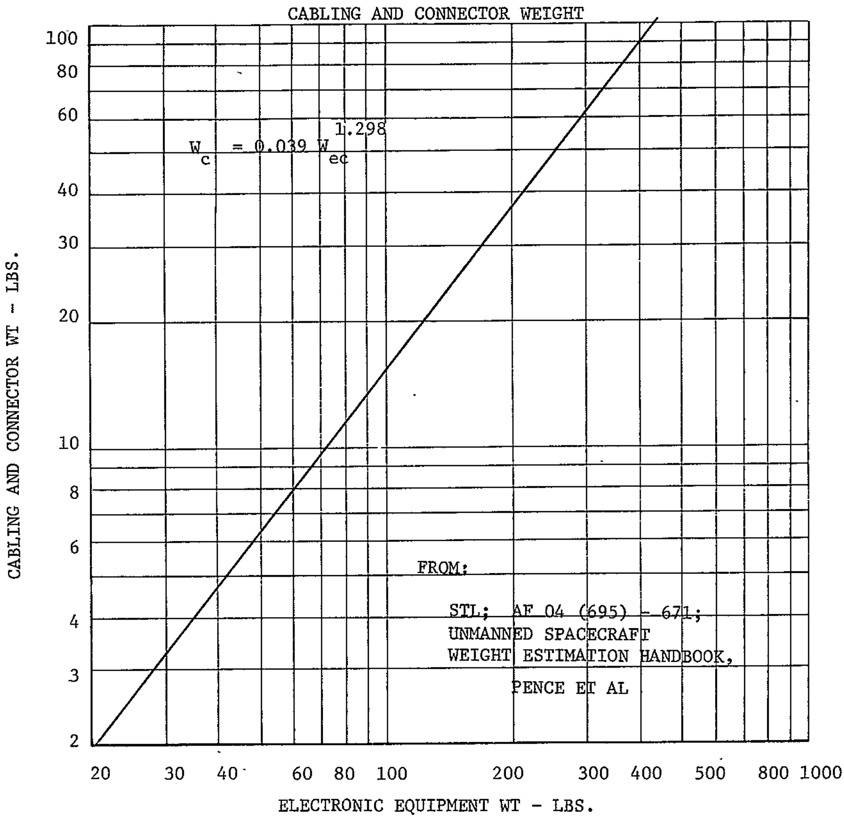

GENERAL EQUATION FOR COMPONENT WEIGHTS
WGN, WIN, WCOM
= COMP Wt. + .412 (COMP WT) .65 +
.039 (COMP WT) 1.298
.412 (COMP WT) .65 = MOUNTING PROVISIONS
.039 (COMP WT) 1.298 = WIRE WT.
SYSTEM WEIGHT - LBS
COMPONENT WEIGHT - LBS
1000
800
600
400
300
200
100
80
60
40
30
20
30
40
50
60
80
100
200
300
400
600
800
1000

FIGURE 6.10-4


CABLING AND CONNECTOR WEIGHT
$W_c = 0.039\, W_{ec}^{1.298}$
FROM:
STL; AF 04 (695) - 671;
UNMANNED SPACECRAFT
WEIGHT ESTIMATION HANDBOOK,
PENCE ET AL
CABLING AND CONNECTOR WT - LBS.
100
80
60
40
30
20
10
8
6
4
3
2
20
30
40
60
80
100
200
300
400
500
800
1000
ELECTRONIC EQUIPMENT WT - LBS.

Eq. 8: WGN = DM(23)+.412*DM(23)**.65+.039*DM(23)**1.298

DM(23) = WEIGHT OF GUID. AND NAV. COMPONENTS IN C.M. [LBS.]

6.10.1.9 Equation 9 defines the weight of the guidance and navigation system in the M.M.

Eq. 9: WGNA = DM(24)+.412*DM(24)**.65+.039*DM(24)**1.298

DM(24) = WEIGHT OF GUID. & NAV. COMPONENTS IN M.M. [LBS.]

6.10.1.10 Equation 10 defines the weight of the on-board checkout system in the C.M.

Eq. 10: WIN = DM(25)+.412*DM(25)**.65+.039*DM(25)**1.298

DM(25) = WEIGHT OF INSTRUMENTATION COMPONENTS IN THE CREW MODULE [LBS.]

6.10.1.11 Equation 11 defines the weight of the on-board checkout system in the mission module.

Eq. 11: WINA = DM(26)+.412* DM (26)**.65+.039*DM(26)**1.298

DM(26) = WEIGHT OF INSTRUMENTATION COMPONENTS IN THE MISSION MODULE [LBS.]

6.10.1.12 Equation 12 defines the weight of the communication system in the crew module.

Eq. 12: WCOM = DM(27)+.412*DM(27)**.65+.039*DM(27)**1.298

DM(27) = WEIGHT OF COMMUNICATION COMPONENTS IN THE CREW MODULE [LBS.]

6.10.1.13 Equation 13 defines the weight of the communication system in the mission module.

Eq. 13: WCOMA = DM(28)+.412*DM(28)**.65+.039*DM(28)**1.298

DM(28) = WEIGHT OF COMMUNICATION COMPONENTS IN THE MISSION MODULE [LBS.]

6.10.1.14 Equation 14 defines the weight of the crew station controls systems in the crew module.

Eq. 14: WCSC = DM(29)+.412*DM(29)**.65+.039*DM(29)**1.298

DM(29) = WEIGHT OF CREW STATION CONTROLS COMPONENTS IN CREW MODULE

6.10.1.15 Equation 15 defin^^ the weight of the crew station control systems in the mission module.

Eq. 15: WCSCA = DM(30)+.412*DM(30)**.65+.039*DM(30)**1.298

DM(30) = WEIGHT OF CREW STATION CONTROL COMPONENTS IN MISSION MODULE [LBS.]

6.10.1.16 Equation 16 defines the weight of the range safety equipment in mission module.

Eq. 16: WRSFTA = DM(31)

6.10.2 Flow Diagram - The logic flow of this subroutine is shown in Figure 6.10-5.

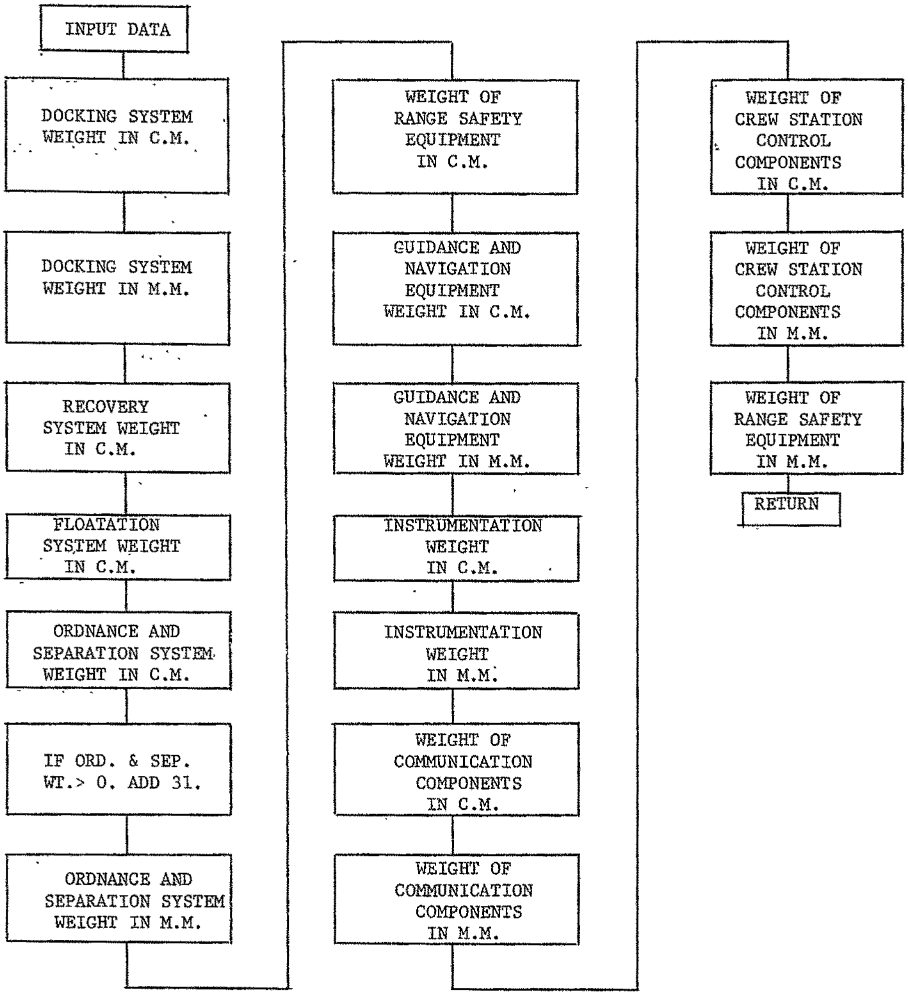
6.11 Mass Properties Subroutine - The Mass Properties Subroutine (MASPR) totals all weight items calculated in other routines, calculates center of gravities (longitudinal and vertical), moments of inertia (yaw and roll), radius of gyration for each inertia, and product of inertia (roll-yaw) for the eight following points in the mission:

1. Launch
2. Effective Launch
3. Injection
4. Docking
5. Pre-Retrograde
6. Reentry
7. Landing
8. Abort

Additionally the subroutine computes the total volume requirement for internally located equipment which is used in the sizing (SIZE) subroutine to compute a spacecraft and mission module length. In order to accomplish these tasks the routine places the homogeneous volumes into the vehicle using an input order for the volume arrangement with the possibility of locating seven items at a specified percent of vehicle length or specified number of feet from the nose tip. The routine calculates the ballast weight required to achieve a specified (by input) reentry center-of-gravity. The subroutine has a built-in iteration loop that converges to a ballast weight within one-tenth of one percent of the previous ballast weight. (The reason for the loop is that ballast volume and hence component c.g.'s are a function of the ballast weight.) A flow diagram for the Mass Properties subroutine is included.

Note that the Mass Properties subroutine is actually composed of 4 routines - the primary routine (MASPR); a routine to order the volumes and establish forward and aft fuselage stations (XLIM); a routine to calculate the vertical center of gravity of each of the homogeneous volumes (ZFIT); and a routine to calculate the forward and aft radius of the idealized cone-frustum shaped homogeneous volume (RFIT).

MISCELLANEOUS SUBROUTINE FLOW DIAGRAM FIGURE 6.10-5



6.11.1 Program Equations - The primary program is broken into five distinct parts: 1) loading into the Mass Properties matrix (MP) the constant-weight-through-the-mission items and deletion of expendables in the entry module; 2) mission module sequencing for retro and entry; 3) volume, c.g., X forward, X aft, radius forward and aft, and the radii of gyration for the elements of the entry module; 4) volume, c.g., X forward, X aft, radius forward and aft, and the radii of gyration for the elements of the mission module; and 5) mission history mass property determination.

The routine intially fills out the MP matrix. This includes such items as cargo transfer at docking, structural cooling water transfer to the outer mold prior to entry, deletion of attitude propellant prior to docking, deletion of environmental oxygen, and jettisoning of tip tanks and aero-fairings.

The second portion of the routine calculates items of Mission Module sequencing prior to retro and during entry. During the entry portion in the mission 20 percent of all ablation material is deleted and 70 percent of all structural cooling water. While the above percentages will vary from trajectory to trajectory they are typical enough that the effects on landing weight will be consistent with assumptions made elsewhere in the program.

Mission Sequencing during the retro phase of the mission is handled by a retro-indicator (RETIND, DMP (72)) which has a value from 1 to 5 corresponding to the 5 modes of retrograding discussed below. The program calculation always follows the entry module plus contiguous portions of the mission module.

1. Integral Case - No weight is in the mission module. The program deletes all mission module structure prior to injection and follows that shape through landing.

2. Modular Case - Mission module contains weight including one retro system. The retrograde system for the reentry module is in the M.M. cone. The cylinder, trailing cone, and portion of the forward cone (excess after enclosing the retro system volume) is separated prior to retro and burns after orbit decay and entry. The retrograding of the R.M. occurs and then the remaining portion of the forward cone is separated from the R.M. The cone also burns after entry. No provision is made for separately retrograding the M.M.

3. Modular Case - Mission Module contains weight including two retrograde systems. One system is in the M.M. forward cone and is for retrograding the R.M. The second system is in the cylinder and retrogrades only the M.M. (less R.M. retrograde system and enclosing structure which are in the forward cone). The M.M. retrograde weight is computed outside the flow of the reentry module logic. Both the M.M. less part of the forward cone and the forward cone burn after entry.

4. Modular Case - Mission Module contains weight including one retrograde system to retrograde the entire M.M. the R.M. is retrograded by a self-contained system. The M.M. retro weight is computed and stored in a slot outside the area for entry module computations.

5. Modular Case - Mission Module contains weight including one retro system for retrograding the entire spacecraft. Separation occurs after retrograding. Mission Module retro weight is not computed as a separate item.

The third portion of the primary program calculates volumes of the internal systems and locates their centers-of-gravity and radius of gyration. Both items are required in the final set of mass properties equations. Also the c.g's are used along with structural c.g's to calculate ballast weight. The computations begin with the calculation of the volumes for each system. This is done by dividing the system weight by a density and multiplying the result by a packaging factor:

Vol = (Weight/Density) Packaging Factor

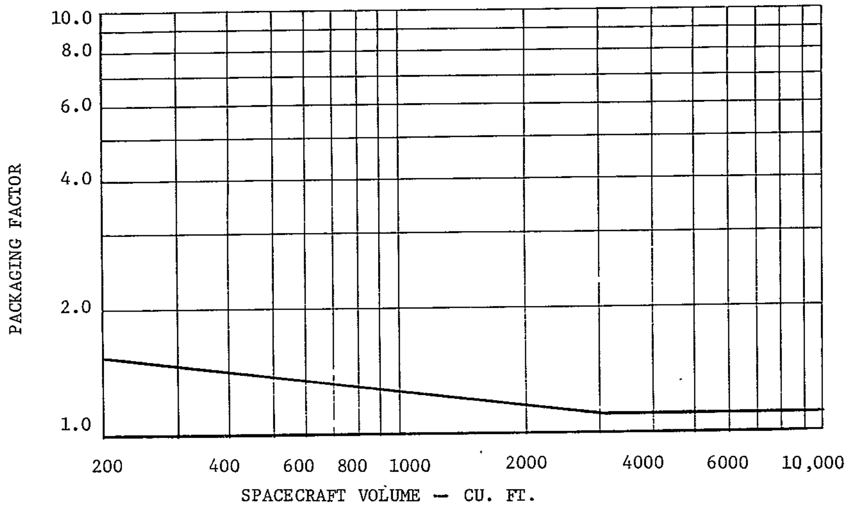
The densities are input (DMP (1) through DMP (20)) as established by experience or ground rule, and the packaging factor is the ratio of installed volume to component volume. The packaging factor (DMP (73)) may be input as a constant (say 1.6) and this value will be used throughout the computations, or by inputting a zero (0.) the packaging factor will be calculated according to the available volume in the spacecraft. The relationship of packaging factor versus volume is shown in Figure 6.11-1. After all volumes have been calculated (cargo volume being defined bv the larger of either launch or return cargo), subroutine XLIM is called to define forward and aft limits for the individual volumes. Each volume (subsystem) is assigned a sequence number (DMP (21) through DMP (40)) to order the volumes from nose tip back.

The ordering of systems may be overridden for the following items:

1. Ballast
2. Personnel
3. Cargo

FIGURE 6.11-1

PACKAGING FACTOR VS. SPACECRAFT VOLUME



4. Nose Gear
5. Main Gear
6. Main Propulsion less engines
7. Main Propulsion engines

When this technique is used DMP (41) through DMP (47) will be used as either a decimal fraction of the length of the vehicle (i.e. .68) or if the input is greater than 1.0 it will be intepreted as feet from the nose tip. (In using the latter input you should know how long your vehicle is going to be to avoid the problem of having systems hanging out the end of the vehicle.) If the overridding technique is used to place a system in the vehicle, then that systems matrix number must fall at (or as close to as possible) the end of the input which ordered the systems (DMP (21) to DMP (40)).

The logic for subroutine XLIM is discussed below. Basically an accumulative available volume plot versus longitudinal body station is generated and used to place systems within the vehicle. Straight line segments are used between 6 finite points. The computations begin by deleting volume for the systems which have a specified c.g. (DMP (41) through DMP (47)). This is shown by dotted lines in

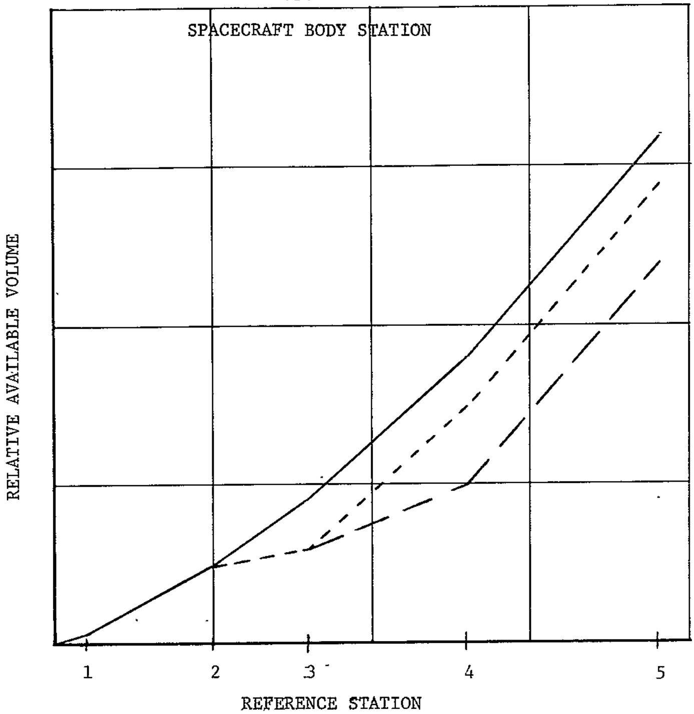
Figure 6.11-2 where two systems have specified c.g.'s - the first falling between sections 2 and 3 and the second falling between 3 and 4. Notice how the effect of deleting (or reserving) volume actually changes the entire curve past the point where the volume is reserved. Two logic checks are made as the new available volume curve is generated: 1) is adequate volume available at the section to meet the input c.g.? - if it is not available the system (or volume) is placed in the vehicle at the first opportunity; 2) is the slope of the accumulative volume curve ever negative? - if it is the volume forward of the negative slope is reduced so that the slope is zero in that section (all the volume in the section is reserved and no additional systems may be placed in the section). After providing volume for the special systems all other systems are placed in the vehicle using the revised accumulative volume plot in the order specified by input. The logic for this is to begin with a zero volume find the forward station, add the volume of the first system and find the aft station for the system, proceed to the next system to find the forward station, add the second system volume to the first volume, find the aft station of the second system etc. All of these computations use the modified available volume curve. If the vehicle length is not adequate to fit in all systems the forward and aft stations of the latter systems are determined using the final slope of the length versus volume curve. Forward and aft limits for all fixed location systems are then established based upon input data (such as radiation, meteorite protection, docking, and abort systems) or upon geometric properties of the vehicle (such as structural shell, fixed and movable aerosurfaces). Next, longitudinal c.g.'s as measured from the nose tip are calculated for all items. For the internal items this is done by first calculating the theoretical radius at the forward and aft stations for the system. (Again a straight-line interpolation is used in RFIT where a plot of radius versus station is generated.) The c.g. for the homogeneous volume is then given by

$$X_{cg} = X_{aft} - \frac{(X_{aft} - X_{fwd})\ (R^2_{aft} + 2.R_{aft}{*}R_{fwd} + 3.R^2_{fwd})}{4.(R^2_{aft} + 2R_{aft}{*}R_{fwd} + R^2_{fwd})} \qquad \text{C.11.1-1}$$

And finally the vertical c.g's are calculated for the internal systems based upon a straight-line interpolation technique. The vertical c.g. is calculated at the longitudinal c.g. for the given system. (This is done in ZFIT.)

FIGURE 6.11-2

AVAILABLE VOLUME
VS.
SPACECRAFT BODY STATION



As the final step in completing the data necessary for mass property determination, the radius of gyration for each of the components is approximated. A number of assumptions were made in this determination.

1. Variable geometry wing radius of gyration is taken as .289 times the span and .289 times the average of root chord and tip chord. At wing deployment the radii change from pitch and roll to roll and pitch.

2. Aerodynamic surfaces are assumed square with the radius of gyration being equal to .289 time square root of the area.

3. Docking system is assumed to be packaged as a homogeneous cube having a density of 40 pounds per cubic foot and therefore the radius of gyration is .408 times the cube root of system weight divided by 40.

4. Tip tank radius of gyration is based upon the cylindrical shell inertia plus eighty percent of the propellant inertia.

The radius of gyration for the internal equipment was based upon the assumption that the equipment is packaged as a homogeneous cone frustum having a small radius SR and a large radius CR. The length of the cone frustum is FL. The computations then proceed as shown below:

$$XB = \left(\frac{FL}{4.}\right) \left(\frac{CR^2 + 2.*CR*SR + 3.*SR^2}{CR^2 + CR*SR + SR^2}\right) \qquad 6.11.1\text{-}2$$

$$CVT = \left(\frac{CR^2 + 3.*CR*SR + 6.*SR^2}{CR^2 + CR*SR + SR^2}\right) \qquad 6.11.1\text{-}3$$

$$SVT = \frac{(CR^5 - SR^5)}{(CR^3 - SR^3)} \qquad 6.11.1\text{-}4$$

$$YYI = \left(\frac{FL^2}{10.}\right) *CVT + \left(\frac{3}{20}\right) *SVT - XB^2 \qquad 6.11.1\text{-}5$$

$$K_{pitch} = \sqrt{YYI} \qquad 6.11.1\text{-}6$$

$$K_{roll} = \sqrt{\left(\frac{3}{10}\right) *SVT} \qquad 6.11.1\text{-}7$$

The yaw radius of gyration is identical to the pitch radius of gyration for the idealized model.

Computations for the mission module (forward cone, cylinder and trailing cone) proceed along similar lines as those described above for the entry module. At the conclusion of mission module calculations, the entry module is ballasted to attain a vehicle center of gravity for aerodynamic stability during entry. The required center of gravity is input as a percent of the vehicle length. The equation for ballast is

BAL = (SXWU - XREQ * SWU)/(XREQ - MP (1,9)) 6.11.1-8

where

BAL is the ballast weight in pounds.

SXWU is the sum of component weight times component center of gravity.

XREQ is the required vehicle center of gravity

SWU is the sum of all component weights

MP (1, 9) is the center of gravity for ballast

The equation will work for nose ballast, in the case of the lifting body vehicles, or for aft ballast, in the case of the ballistic vehicle. Negative ballast is not allowed. If in the lifting body case the actual center of gravity is forward of the required (by input) center of gravity the ballast weight is set equal to zero. Similarly if the actual center of gravity is aft of the required center of gravity in the ballistic case (assuming the ballast location in the ballistic case has been specified in the aft part of the vehicle), the final ballast weight has converged to within one-tenth of one percent of itself on two successive passes through the equations. Completion of the ballast calculation then allows a solution for mass properties.

The mass property determination uses the standard equations as shown below.

$$TGW = W_{total} = \Sigma\, W_i \qquad 6.11.1\text{-}9$$

$$TGWX = \Sigma\, W_i X_i \qquad 6.11.1\text{-}10$$

$$TGWZ = \Sigma\, W_i X_i \qquad 6.11.1\text{-}11$$

$$TWXZ = \Sigma\, W_i X_i Z_i \qquad 6.11.1\text{-}12$$

$$TWXX = \Sigma\, W_i X_i X_i \qquad 6.11.1\text{-}13$$

$$TWZZ = \Sigma\ W_i Z_i Z_i \qquad 6.11.1\text{-}14$$

$$TWKXX = \Sigma\ W_i K_{xi} K_{xi} \qquad 6.11.1\text{-}15$$

$$TWKYY = \Sigma\ W_i K_{yi} K_{yi} \qquad 6.11.1\text{-}16$$

$$TWKZZ = TWKYY \qquad 6.11.1\text{-}17$$

With these definitions the elements in the mass property array can be evaluated.

Total weight at Mission Point J:

$$TGW \qquad 6.11.1\text{-}18$$

Longitudinal center of gravity for Mission Point J:

$$(XBAR=)\ TGWX/TGW \qquad 6.11.1\text{-}19$$

Vertical center of gravity at Mission Point J:

$$(ZBAR=)\ TGWZ/TGW \qquad 6.11.1\text{-}20$$

Inertia about pitch axis at Mission Point J:

$$\frac{(TWXX-TGW*XBAR^2 + TWZZ-TGW*ZBAR^2 + TWKYY)}{32.174} \qquad 6.11.1\text{-}21$$

Inertia about roll axis at Mission Point J:

$$\frac{(TWZZ-TGW*ZBAR^2 + TWKXX)}{32.174} \qquad 6.11.1\text{-}22$$

Inertia about yaw axis at Mission Point J:

$$\frac{(TWXX - TGW*XBAR^2 + TWKZZ)}{32.174} \qquad 6.11.1\text{-}23$$

Product of inertia in Roll-Yaw plane:

$$\frac{(TWXZ - TGW*XBAR*ZBAR)}{32.174} \qquad 6.11.1\text{-}24$$

Radius of gyration for pitch at Mission Point J:

$$5.67217 \sqrt{\text{pitch inertia/TGW}} \qquad 6.11.1\text{-}25$$

Radius of gyration for roll inertia at Mission Point J:

$$5.67217 \sqrt{\text{roll inertia/TGW}} \qquad 6.11.1\text{-}26$$

Radius of gyration for yaw inertia at Mission Point J:

$$5.67217 \sqrt{\text{yaw inertia/TGW}} \qquad 6.11.1\text{-}27$$

Notice that in the computations several basic assumptions were made.

1. roll inertia $\left.\begin{array}{l} \text{YBAR} = 0. \\ \Sigma W_i Y_i = 0. \end{array}\right\}$ no lateral offset

2. yaw inertia $\left.\begin{array}{l} \text{YBAR} = 0. \\ \Sigma W_i Y_i = 0. \end{array}\right\}$ no lateral offset

   TWKZZ = TWKYY

3. Pitch - Yaw product = 0.

4. Pitch-roll product = 0.

5. Conversion factors are for input units in pounds and feet to obtain slug-ft^2.

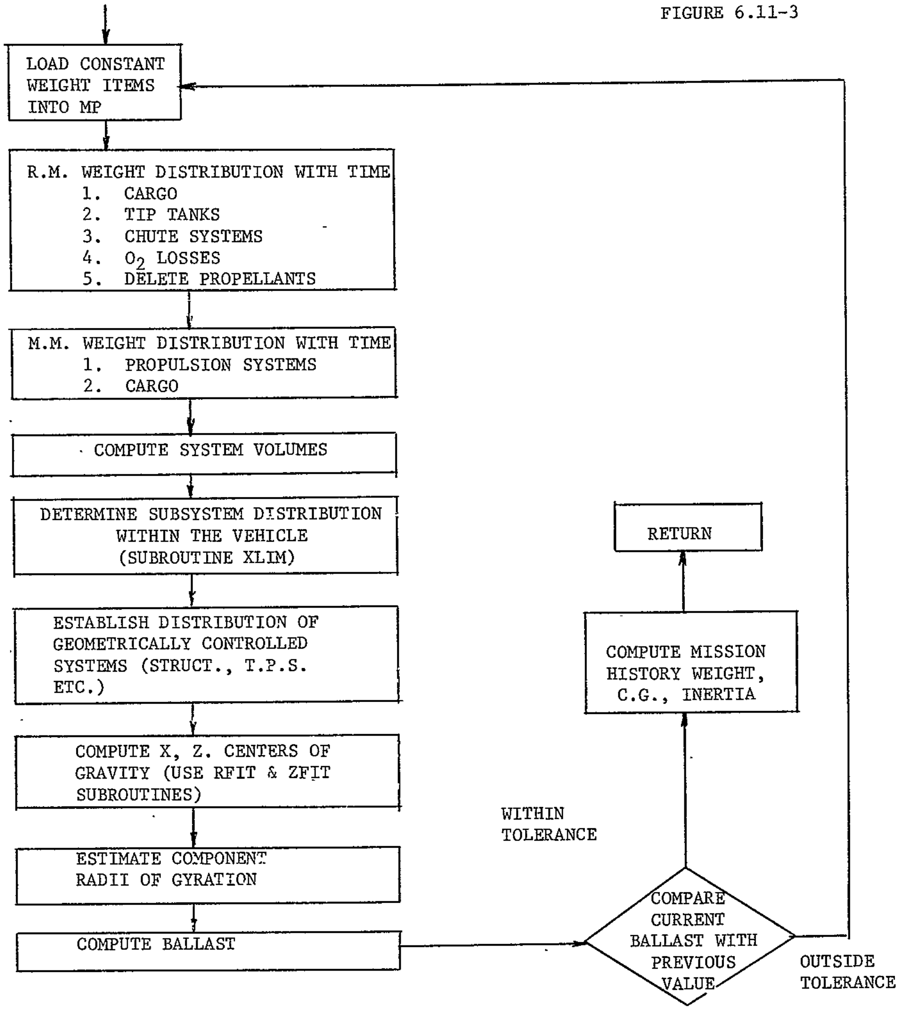
6.11.2 Flow Diagram - The logic flow for this subroutine is shown in Figure 6.11-3.

6.12 Loads Subroutine - Only two conditions are investigated for loads analysis; max alpha-q and landing. The maximum angle of attack/dynamic pressure condition usually results in the maximum loads on the aft section of the spacecraft. The landing condition usually results in the maximum loads in the forward and middle sections if the vehicle is a land recoverable lifting body shape.

6.12.1 The subroutine first computes the maximum alpha-q load condition to determine the axial and bending loads on the entry vehicle at the interface with the cargo/propulsion module or the booster. It also computes the loading on the

MASPR FLOW DIAGRAM

FIGURE 6.11-3



cargo/propulsion module for this condition. Simple adapters are sized by aerodynamic loading only.

The loading conditions for tip tanks and their attachment are next considered and, finally the landing loads are computed. All loadings are compared and the maximum loading condition designs the body structure.

6.12.1.1 Maximum Alpha-q Condition - The maximum alpha-q condition is evaluated from fixed alpha-q/nz data.

6.12.1.2 Adapter Aerodynamics - The spacecraft to launch vehicle adapter aerodynamics are from two sources. For the conical sections of the adapter the pressure data are from NACA TR 1135 for a Mach Number of 1.4. The cylinderical section airloads are based on the information found in SSD-CR 63-171.

6.12.1.3 Tip Tank Aerodynamics - The tip tank aerodynamics present a complicated problem. A much simplified approach is used which should be indicative of the use of tip tanks. The normal force slope coefficient and axial force coefficient for the sum of all tip tanks is assumed to be some factor times the core spacecraft normal and axial force coefficients. The center of pressure to length ratio is assumed to be some factor of the core center of pressure to length ratio. Therefore, the tip tank data are all ratios of the core data and allow variation of the tip tank sizes without changing the data.

6.12.1.4 Tip Tank Structural Attach Points - The structural attach points of these tip tanks is assumed to be at two points. The aft one is near the rear of the core spacecraft and it carries all of the axial load and the beamed out portion of the normal load. The forward attach point is at 25% of the tip tank length unless that point is farther forward than the first section cut of the core spacecraft aft of the nose. If the latter is true, it is placed at that section cut.

6.12.1.5 Launch Escape System - The Launch Escape System (LES), if used, is a generalized form based on the maximum cross sectional area of the spacecraft recovery pod. The aerodynamic data are assumed to be

$$Cna = .00607 \text{ per degree}$$

$$Ca = .1167$$

with the center of pressure at 82.5% of the heat shield diameter forward of the nose of the spacecraft.

6.12.1.6 Landing Loads - Of the two major conditions at landing, the forward gear impact condition is usually the worst for structural loads. The forward gear loads are evaluated from

$$F = \frac{E}{S*n} \qquad 6.12.1\text{-}1$$

where F is the force normal to the surface

E is the gear energy

S is the gear stroke

and n is the energy absorber efficiency.

The gear energy requirements are evaluated from

$$E = Me/2*((zd + td*A)^2 - 2*g*S) \qquad 6.12.1\text{-}2$$

where zd is the sink speed at touchdown

td is the pitch rate at touchdown

A is the horizontal distance from the CG to the gear contact point

and Me is the effective mass over the gear.

The effective mass is derived from

$$Me = 1/(1/M + A*(A + u*h)/I) \qquad 6.12.1\text{-}3$$

where M is the total spacecraft mass

I is the total spacecraft pitch moment of inertia

u is the contact coefficient of friction

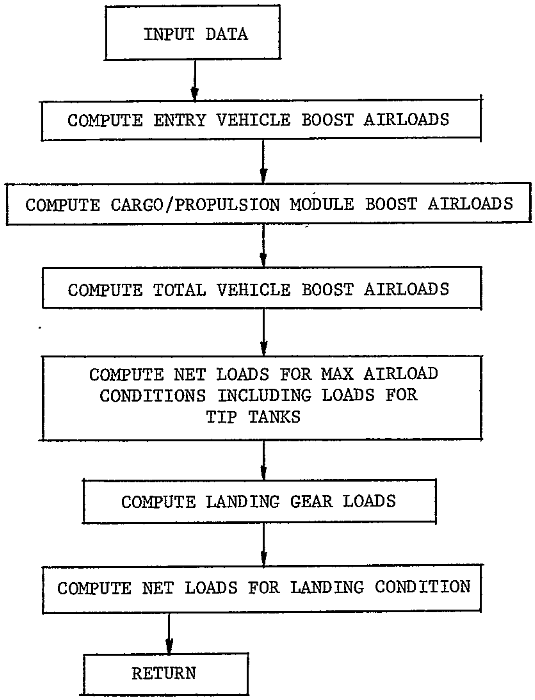
and h is the vertical height of the CG above the surface.

6.12.2 Flow Diagram - The logic flow of this subroutine is shown in Figure 6.12-1.

6.13 Structures Subroutine - In advanced design studies, it is not practical to analyze each structural item in sufficient depth to determine its associated weight. Therefore, weight estimation models, correlated where possible to hardware, are required. These models are not intended to yield optimized structural designs. They must, however, provide data adequate to define reasonable weights and their sensitivities to the design and performance criteria applicable to each advanced design study. Two approaches are employed for structural sizing: 1) Analytical equations and 2) Semi-empirical equations. Effort was directed toward obtaining reasonable analytical weight prediction equations for major structural items where

FIGURE 6.12-1

LOADS SUBROUTINE FLOW DIAGRAM



statistical data was limited. These major items are: primary shell, primary frames (rings) and structural thermal protection system. The latter item is documented in Section 6.17. For remaining structural items, semi-empirical equations are obtained from established statistical data compiled from existing hardware as previously discussed. Table 6.13-1 summarizes structural items requiring weight estimates in spacecraft advanced design studies. Capabilities and limitations of each model is discussed. Engineering flow diagrams are included as an aid in understanding the logic of each sizing model.

SYMBOLS

F_{tu}	Ultimate tensile strength, psi
F_{cy}	Compressive yield strength, psi.
E	Young's modulus of elasticity, psi
F_{cc}	Crippling stress, psi
F_c	Allowable column stress, psi
F_b	Allowable bending stress, psi
F_{rb}	Reference bending stress employed in plastic bending analysis, psi
f_c	Applied, ultimate compressive stress, psi
f_t	Applied, ultimate tensile stress, psi
P	Applied, ultimate axial load, pounds
P_c	Allowable column load per Johnson's equation, pounds
P_{cr}	Critical column load per Euler's equation, pounds
P_{all}	Axial load at Failure, pounds
q	Applied, ultimate uniform radial loading, pounds per inch
q_t	Applied, ultimate tensile uniform radial loading, pounds per inch
q_{cr}	Critical uniform radial loading, pounds per inch
p	Applied, ultimate pressure, psi
M	Applied, ultimate bending moment, inch-pounds
M_o	Bending moment due to lateral load and corresponding to y_o, inch-pounds
M_{all}	Allowable bending moment in plastic bending, inch-pounds

TABLE 6.13-1

STRUCTURAL ITEMS

Structural Sizing Based on Analytical Equations	Structural Sizing Based on Semi-Empirical Equations Correlated to Hardware		
1. Innerbody Shell: Circular Non-circular/Flat sides	1. Control Surfaces: Leading Edges Trailing Edges Basic Structure	6. Engine Thrust Structure	13. Aerodynamic Fairing
2. Frames (Rings): Circular Non-circular	2. Nose Tip and Leading Edge Structure	7. Access/Egress Provisions: Structural Non-structural	14. Abort Tower Structure
3. Structural Heat Protection: Shingles	3. Variable Geometry Wing Structure: Basic Wing Carry-thru Rotation Mechanism	8. Windows 9. Flooring	15. Meteorite Protection 16. Radiation Protection
	4. Bulkheads: Compression Tension	10. Landing Gears: Ballistics Conventional	17. Separation Provisions: Manufacturing Splice Access Splice
	5. Support Structure: Attachments Truss/Beams	11. Docking Structure 12. Tankage: Gaseous Supercritical	18. Miscellaneous Mounting Provisions

y	Maximum final or total deflection, inches
y_o	Maximum deflection due to lateral and/or moment plus initial eccentricity, inches
α	Ratio of P/Pcr or P_{all}/Pcr, scalar
$\frac{1}{1-\alpha}$	Deflection magnification factor, scalar
A	Cross-sectional area of beam or column, square inches
L	Beam-column length, spacing between frames, inches
L'	Effective beam-column length, $L'=L/\sqrt{c}$, inches
t	Thickness, inches
b	Element width used in crippling equations, inches
h	Frame web depth, inches
s	Corrugation pitch, inches
I	Area moment of inertia, inches ^{4}
ρ	Radius of gyration or distance from center of hoop tension circle to periphery of frame, inches; material *
c	Fixity coefficient, scalar; distance to extreme fiber, inches
Q	Static moment of area about neutral axis for plastic bending, inches ^{3}
R	Radius of cylindrical shell or frame or radius of hoop tension circle, inches
Z	Section modulus, inches ^{3}
$\overline{W}$	Unit weight, pounds per square foot

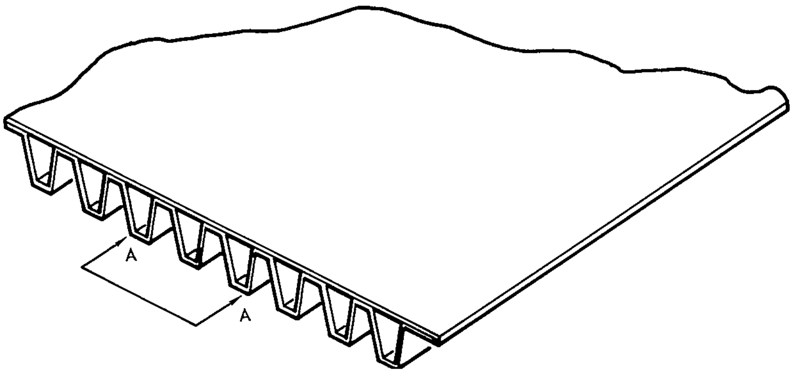
* density, pounds per cubic inch

6.13.1 Structural Shell - Structural shells, as considered in the following context, along with intermediate frames make-up the basic structural airframe. The shell model developed for all spacecrafts lies along an inner structural mold-line which is concentric with the external mold-line. The spacing between the two mold-lines is dictated chiefly by the frames which are attached external to the structural shell providing it with lateral strengths and stiffness and also support for the heat protection shingles.

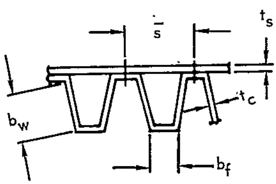
Spacecrafts have inner body structural shells which vary from cylindrical or conical cross-sections to flat sided, non-circular cross sections. It is necessary to consider a structural concept that can accommodate all of these shapes and still

FIGURE 6.13-1

SINGLE-SKIN, TRAPEZODIAL CORRUGATION GEOMETRY
(SIZING MODEL)



$t_s = 0.488\ \bar{t}$
$s = 20\ t_s$
$t_c = 0.5\ t_s$
$b_f = 6.0\ t_s$
$b_w = 15.0\ t_s$



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provide adequate strength and stiffness during all mission phases. Two basic requirements of the innerbody shell structure are to provide load paths for carrying aerodynamic and inertia induced loads (i.e. body bending moments, axial loads and shear loads) and to provide a pressure shell for carrying internal pressures. When this latter condition occurs simultaneously with aerodynamic and inertia loads, the structural shell is analyzed using beam-column analysis.

The analytical equations are developed for a single skin, stiffened longitudinally with open-face trapezoidal corrugations as illustrated in Figure 6.13-1. However, three different concepts are considered as discussed and illustrated in Volume II, Book 1, Section 6.2.2. Correlation factors between these three concepts and the analytical equations are developed in Section 6.13.2.

6.13.1.1 Non-Circular Shells - Single skin, open-face corrugations are analyzed as though each pitch of skin-corrugation acts as an individual beam-column. Beam-column length is equal to the interval between frames. Boundary conditions consider: (1) the primary shell skin to exhibit negligible hoop tension or compression capability; therefore, each pitch of skin-corrugation beams the entire lateral pressure loads to structural frames, and (2) end fixity of the beam-column to be fully fixed. Extent of conservatism exhibited by assuming negligible hoop capabilities is highly dependent upon shell deviation from a cylindrical cross-section. Cylindrical shells provide excellent load paths for carrying hoop loads.

The following work develops the shell sizing model which is taken directly from Reference 6.13-1.

Beam Columns - A beam-column is a compression member which is also subjected to bending loads. Bending may be caused by eccentric application of the member, lateral loadings, or a combination of these causes.

The final deflection at the center is:

$$y = y_o \left(\frac{1}{1-\alpha}\right).$$

The moment on the column can then be expressed by the following equation:

$$M = Py + M_o$$

The margin of safety of a beam-column can be calculated using the following interaction formula:

$$R_b + R_c = 1, \text{ where}$$

$$R_b = \frac{P_{all}\left(\frac{y_o}{1-\alpha}\right) + M_o}{M_{all}} \quad \text{and}$$

$$R_c = \frac{P_{all}}{F_{cc}A}$$

Substituting the ratios, R_b and R_c into the interaction formula yields the following equation:

$$\left(\frac{P_{cr}}{F_{cc}A}\right)\left(\frac{P_{all}}{P_{cr}}\right)^2 - \left(\frac{P_{cr}}{F_{cc}A} + \frac{P_{cr}y_o}{M_{all}} - \frac{M_o}{M_{all}} + 1\right)\left(\frac{P_{all}}{P_{cr}}\right) - \frac{M_o}{M_{all}} + 1 = 0.$$

This is a quadratic equation with P_{all}/P_{cr} as the unknown. It can be solved by the usual quadratic formula:

$$\frac{P_{all}}{P_{cr}} = \frac{-b \pm \sqrt{b^2-4ac}}{2a} \quad , \text{ where}$$

$$a = \frac{P_{cr}}{F_{cc}A}$$

$$b = -\left(\frac{P_{cr}}{F_{cc}A} + \frac{P_{cr}y_o}{M_{all}} - \frac{M_o}{M_{all}} + 1\right) \quad \text{and}$$

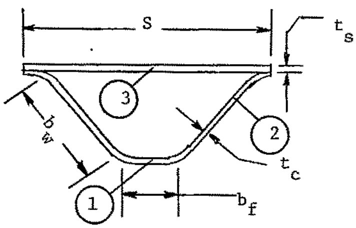
$$c = -\frac{M_o}{M_{all}} + 1$$

The negative sign before the radical gives the proper answer when substituted into the following expression:

$$P_{all} = P_{cr}\left(\frac{P_{all}}{P_{cr}}\right)$$

Before proceeding, it is necessary to solve for the quadratic equation coefficients, a, b and c in terms of the single-skin, open face corrugation geometry. This is accomplished in five basic steps:

1) Determination of Allowable Crippling stress, F_{cc}. To determine allowable crippling stress the following geometric relationships are used:



$t_s = .05s$ $\qquad$ $A = .1025s^2$

$t_c = .025s$ $\qquad$ $I = .0074s^4$

$b_f = .30s$

$b_w = .75s$

From Reference 6.13-1, the crippling stress for a no edge free element may be founc from the equation:

$$F_{cc} = 1.41\left(\frac{F_{cy}}{E}\right)^{.595}\frac{E}{\left(\frac{b}{t}\right).81}.$$

For a given element, the maximum value allowed for F_{cc} is F_{tu}. In any equation used for determining the composite crippling stress of a skin-corrugation, it is necessary to account for three geometrically different elements as follows:

Element 1, where $\frac{b_f}{t_c} = \frac{.30s}{.025s} = 12,$

Element 2 , where $\frac{b_w}{t_c} = \frac{.75s}{.025s} = 30$, and

Element 3 , where $\frac{s}{t_s} = \frac{1.s}{.05s} = 20$.

The following combinations of conditions are possible stress levels for the skin-corrugation elements:

a) No elements working at F_{tu},

b) One element working at F_{tu},

c) Two elements working at F_{tu}, and

d) All three elements working at F_{tu}.

For each possible combination, a composite F_{cc} of skin-corrugation is obtained from the following equations shown in Table 6.13-2.

Table 6.13-2

Define $Z = \left(\frac{E}{F_{tu}}\right)\left(\frac{F_{cy}}{E}\right)^{.595}$

lower limit Z	upper limit Z	
0	5.3	$\frac{F_{cc}}{F_{tu}} = .1215\ Z$
5.3	8.02	$\frac{F_{cc}}{F_{tu}} = .094\ Z + .146$
8.02	11.11	$\frac{F_{cc}}{F_{tu}} = .0328\ Z + .634$
11.11	∞	$\frac{F_{cc}}{F_{tu}} = 1.$

2) Determination of Allowable Buckling Load, P_c - The allowable buckling load is calculated from Johnson's Parabola for short columns:

$$P_c = F_{cc}A - \frac{F_{cc}^2 (L'/p)^2}{4\pi^2 E} A.$$

For the skin-corrugation structure with a full fixity as the assumed boundary condition, the Johnson's equation becomes:

$$P_c = .1025 L^2 \left[\frac{F_{cc}}{(L/s)^2} - \frac{F_{cc}^2}{11.42 E} \right]$$

Calculating the column stress from Johnson's equation,

$$F_c = \frac{P_c}{A} ,$$

It must be verified that F_c is equal to or greater than one-half of F_{cc}. If this is not the case, then the basic Euler column equation replaces Johnson's equation, or:

$$P_c = \pi^2 \frac{EI}{(L')^2} \quad \text{and } F_c = \frac{P_c}{A}$$

3) Determination of Initial Deflection, y_o, and Initial Moment, M_o - These values are obtained from "Formulas for Stress and Strains", by R. J. Roark, 4th ed., dated 1965, for a fixed end beam carrying a uniform lateral load. The equations, after substituting appropriate geometries, are:

$$y_o = \frac{1}{2.84} \quad \frac{qL^4}{Es^3} \quad \text{and}$$

$$M_o = \frac{1}{24} qsL^2$$

4) Determination of Allowable Bending Moment, M_{all} - In determining the allowable bending moment, the basic plastic bending equation found in 6.13-1 is employed:

$$M_{all} = (Q_1 + Q_2)\ F_{rb}.$$

Substituting the geometric relationship for the skin - corrugation into this expression, the following expression for M_{all} is obtained.

$$M_{all} = .0193s^3\ F_{tu}.$$

5) Quadratic Equation Coefficients - The coefficients a, b and c are expressed as follows:

$$a = \frac{P_c}{F_{cc}A} = \frac{\left[\dfrac{F_{cc}}{(L/s)^2} - \dfrac{F_{cc}^{\ 2}}{11.42E}\right]}{\left[\dfrac{F_{cc}}{(L/s)^2}\right]}$$

$$b = -\ [(a+c) + 8.46\ \frac{L}{E}\ (\frac{P_c}{L^2})\ (\frac{L}{s})^4\ \frac{1}{L}\ (\frac{M_o}{M_{all}})],\ \text{and}$$

$$c = 1 - 2.16\ (\frac{L}{s})^2\ (\frac{q}{F_{tu}})\ .$$

Where, if $F_c \geq 1/2\ F_{cc}$, $P_c = .1025L^2 \left[\dfrac{F_{cc}}{(L/s)^2} - \dfrac{F_{cc}^{\ 2}}{11.42E}\right]$ (Johnson's equation)

If $F_c < 1/2\ F_{cc}$, then $P_c = \pi^2\ \dfrac{EI}{(L')^2}$ (Euler's equation)

The subroutine utilizes an iterative process where values of the skin thickness, t_s, are varied until P_{all} approaches P or the margin of safety approaches zero. As shown on Figure 6.13-1, the value calculated for t_s must satisfy the requirement:

$$\frac{t_s}{2} = t_c \geq t_{minimum}\ .$$

The value, $t_{minimum}$, represents a minimum sheet thickness based upon manufa and/or handling limitations used in the aerospace industry.

6.13.1.2 Circular Shells - As previously discussed, cylindrical and conical shells are capable of carrying hoop loads. However, the present equations do not account for this capability. Therefore, conical shells, which are employed on ballistic vehicles and cargo propulsion modules plus launch vehicle adapters, are conservatively sized by the techniques for non-circular shells.

6.13.2 ALTERNATE CONCEPTS FOR STRUCTURAL SHELLS - As previously discussed, Volume II, Book 1, Section 6.2.2, three methods of construction are considered for the structural shell: (1) skin-stringers with frames, (2) single-skin, open face square corrugations with frames and (3) single-skin with frames. Since the mathematical model developed in Section 6.13.1 is an existing math model and is not included in the above set, there are several procedures which can be initiated. First, new math models can be derived based on analysis methods of Section 6.13.1. Second, correlation factors relating the existing math model to the new methods of construction can be developed. Or, third, a combination of the above two procedures can be used. The last procedure was finally adopted. Rather than developing new math models for concepts one and two, correlation factors were readily obtained. However, for the third concept a new math model was developed.

Correlation Factors - The beam-column method of analysis employed in Section 6.13.1 is used in obtaining correlation factors for the methods of construction: skin-stringers with frames and single-skin, open face, square corrugations with frames. Figures 6.13-2 and 6.13-3 are used to illustrate the procedure for defining correlation factors. Both figures contain plots comparing the structural shell weights for concepts one and two and the existing math model. For a range of representative load intensities (i.e., 400 lbs/inch. to 4000 lbs/inch.), an average correlation factor is computed for each concept. Table 6.13-3 summarizes the correlation factors for four candidate materials for either a pressurized compartment or non-pressurized compartment.

The procedure employed for sizing shell structure is to utilize the existing math model and adjust the shell size for computing structural weight by applying the correlation factors. Figure 6.13-4 outlines this procedure with a

FIGURE 6.13-2

ALUMINUM SHELL WEIGHT

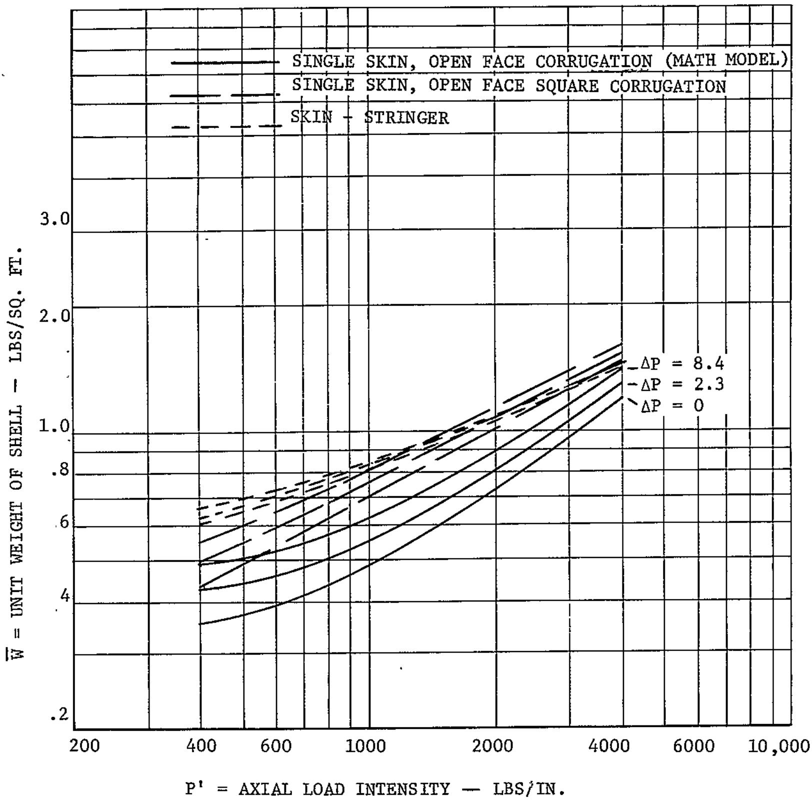
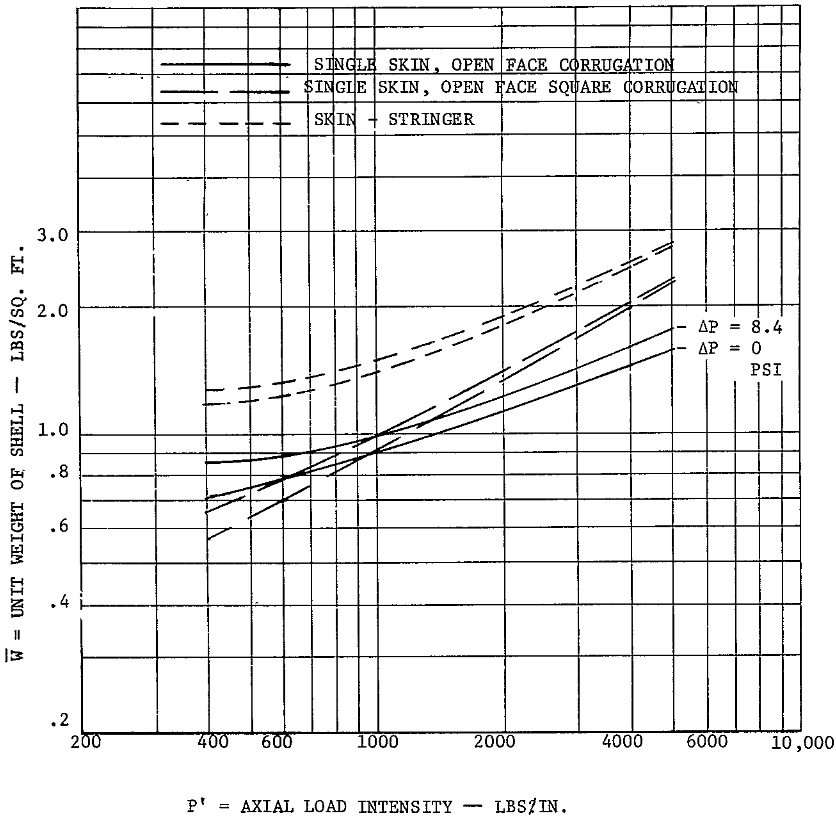


FIGURE 6.13-3

17-7PH STAINLESS STEEL SHELL WEIGHT



flow diagram. This diagram shows correlation factors applied to calculated, equivalent thicknesses where the previous two figures derived these factors by ratioing unit weights. The following equation relates unit weight to equivalent thickness:

$$\overline{W} = 144 \rho \overline{t}$$

It is evident from this equation that the correlation factors can be applied to either equivalent thickness or unit weight.

The following equations relate the equivalent thicknesses to stringers and corrugation thicknesses respectively:

$$(1) \quad \overline{t} = 1.6 \ t, \text{ and}$$

$$(2) \quad \overline{t} = 3. \ t_s$$

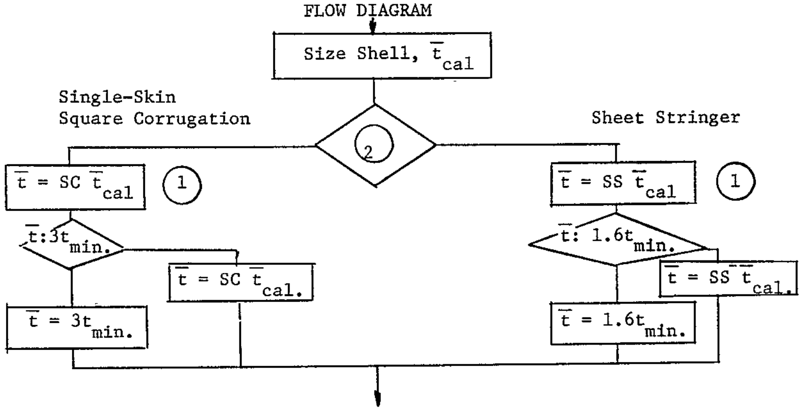
TABLE 6.13-3

CORRELATION FACTORS

Materials \ Methods of Construction	Skin-Stringer		Single-Skin, Open Face Square Corrugation	
	Pressurized Compartment	Unpressurized Compartment	Pressurized Compartment	Unpressurized Compartment
Aluminum (2219-T87)	1.26	1.36	1.26	1.32
Magnesium (HK31A-H24)	1.05	1.13	1.03	1.09
Titanium (8AL-1Mo-1V)	.93	.98	1.01	1.04
Stainless Steel (17-7PH)	1.45	1.51	.97	.99

FIGURE 6.13- 4

FLOW DIAGRAM



Notes

1 SC, and SS are the correlation factors applied to calculated thicknesses of the math model to yield equivalent, load carrying thicknesses for single-skin square corrugation, and sheet stringer respectively.

2 Type of Construction Indicator.

6.13.3 SIZING MODEL FOR STRUCTURAL FRAMES - Structural frames for entry vehicles of this study serve two primary functions. They support thermal protection shingles and provide strength and stiffness to maintain body shape of pressurized compartments.

Sizing models for both circular and non-circular frames assume specific geometric relationships. These relationships and/or assumptions are:

1) Frames utilize channel cross-sections,
2) Web depth to flange width ratio is equal to 3 (i.e., h/b = 3),
3) Channels are of constant thickness,
4) Web depth is a function of frame height plus a constant, and
5) No effective skin of shell structure is used with frames.

The following two sections develop sizing models for circular and irregularly shaped frames. The procedure for sizing frames in entry vehicles and mission module adapters is to subdivide the spacecraft into several sections.

Frame structure is sized at these section cuts and unit weights (pounds per square foot) are obtained. With these weights at two adjacent section cuts, an average value is calculated and used to obtain the total frame weight between the two cuts. Adding up all sections yields the total frame weight for the structure.

6.13.3.1 CIRCULAR FRAMES (RINGS) - Three modes of failure are analyzed when developing a sizing model for circular frames. These are: (1) general instability, (2) local instability (crippling), and (3) hoop tension failure. Condition (1) and (2) are investigated for uniform, external collapse pressures and condition (3) is investigated for uniform, internal burst pressures. Analytical expressions are generated defining required frame thicknesses for each mode of failure. The sizing model examines all three expressions and selects the maximum thickness with a further stipulation that this thickness is equal to or greater than a minimum thickness stipulated by manufacturing and handling limitations.

The critical running load per unit length of circumference for general instability is:

$$q_{cr} = \frac{3EI}{R^3} .$$

Derivation of this equation can be obtained from Reference 6.13-1. Based upon geometric relationships for channel frames, the area moment of inertia is expressed as,

$$I = \left[\frac{1}{2}\left(\frac{b}{h}\right) + \frac{1}{12}\right] h^3 t.$$

Substituting this expression into the initial equation and solving for t, the following equation is obtained:

$$t = \frac{q_{cr} R^3}{3E} \left[\frac{1}{h^3\left\{\frac{1}{2}\left(\frac{b}{h}\right) + \frac{1}{12}\right\}}\right] .$$

Substituting the ratio of web depth to flange width equal to three (i.e., h/b = 3), the equation simplifies to,

$$t = \frac{4}{3} \frac{q_{cr}}{E} \left(\frac{R}{h}\right)^3 .$$

Critical running load, q_{cr}, in the above equation, is equated to the applied running load based on a zero margin of safety or:

$$M.S. = \frac{q_{cr}}{q} - 1 = 0, \; q_{cr} = q \text{ where}$$

$$q = pL.$$

For local instability, an expression is obtained from the non-dimensional crippling curves of Reference 6.13-1. The expression for F_{cc} is:

$$F_{cc} \cong k \frac{\left[F_{cy}\right]^{.6} \left[E\right]^{.4}}{(b/t)^{.8}} .$$

The coefficient, k , depends on boundary conditions: 1) For one edge free (O.E.F.), k = .56 and 2) For no edge free (N.E.F.), k = 1.41.

Utilizing the previously defined frame geometry, the composite crippling stress for the channel frame becomes:

$$F_{cc} = 1.385 \frac{\left[F_{cy}\right]^{.6} \left[E\right]^{.4} \left[t\right]^{.8}}{\left[h\right]^{.8}} .$$

Internal compressive stress for a uniform collapse pressure is:

$$f_c = \frac{qR}{A_f} \text{, where } q = pL.$$

Expressing the frame cross-sectional area, A_f, in terms of its individual elements, the compressive hoop stress becomes:

$$f_c = \frac{3}{5} \frac{qR}{ht} .$$

Proceeding as was done for general instability, the margin of safety is equated to zero which yields the following:

$$f_c = F_{cc}.$$

The stress level must not exceed the ultimate tensile strength, F_{tu}. Substituting the appropriate parameters into the above equation and solving for frame thickness, the frame sizing equation becomes:

$$t = .629 \frac{(qR)^{.555}}{h^{.111} F_{cy}^{.333} E^{.222}} .$$

Internal burst pressures induce hoop tension loads. The internal hoop tension stress is expressed as:

$$f_t = \frac{P}{A_f}\text{, where } P = q_t R \text{ and } q_t = p_t L.$$

Expressing the cross-sectional area, A_f, in terms of individual elements, the tensile hoop stress becomes:

$$f_t = 3/5 \frac{q_t R}{ht} .$$

For a zero margin of safety, the following expression results:

$$f_t = F_{tu}.$$

Solving for frame thickness, the frame sizing equation for hoop tension is:

$$t = \frac{3}{5}\ \frac{q_t R}{hF_{tu}}.$$

6.13.3.2 NON-CIRCULAR FRAMES - Since the ideal shape for pressurized shells is circular, the greater the deviation from a circular shape the greater will be the frame internal bending moment distribution. The method of Reference 6.13-1 relating to irregular shaped pressure vessels is employed for determining internal bending moments. Internal bending moments around the periphery of an irregularly shaped frame may be determined by first locating the center and radius of a circle which has an equivalent hoop tension. This circle, in order to satisfy the laws of static equilibrium, will intersect the frame periphery at inflection points (i.e., zero bending moments). The moment at any point on the periphery of the frame is then a function of the radial ordinates of the irregularly shaped frame centerline and the equivalent circle; and the section of critical moment may be determined visually from a scale drawing. For the present sizing model, the largest internal bending moment is used which exists at either the top, bottom or side of the peripherical frame. The expression for internal bending moment is:

$$M = \frac{1}{2}\, q\ (\rho^2 - R^2), \text{ where } q = pL.$$

Similar to the methodology applied in sizing circular frames, an expression is derived for sizing irregularly shaped, channel frames. Again, Reference 6.13-1 uses an empirical equation which yields the allowable bending stresses for channels subjected to normal loading. This equation is:

$$F_b = \frac{450{,}000}{(b/t)^{.78}} \left(\frac{b}{h}\right)^{.27} \sqrt{\frac{F_{cy}/E}{.0604}}$$

The allowable bending moment is obtained by multiplying this stress by the section modulus or:

$$M_{all.} = F_b Z_f, \text{ where } Z_f = \frac{I_f}{c}.$$

If the margin of safety is equated to zero or:

$$M.S. = \frac{M_{all.}}{M} - 1 = 0,$$

then the following expression results:

$$M_{all.} = M.$$

Substituting the various parameters into this expression and solving for frame thickness, the sizing equation becomes:

$$t = 1.575(10)^{-4}\, M^{.56} \left(\frac{E}{F_{cy}}\right)^{.279} \left(\frac{1}{h}\right)^{.678}$$

One further stipulation is that the allowable bending stress must not exceed the ultimate tensile strength. If in the computations, this situation occurs the sizing model is modified to the following expression:

$$t = 2\,\frac{M}{h^2 F_{tu}}$$

In addition, one further test requires that the calculated thickness must be equal to or greater than the minimum thickness stipulated by manufacturing and handling limitations.

6.13.4 - Miscellaneous Structural Items - Referring to Table 6.13-1 eighteen structural items are listed under Miscellaneous Structural Items. Of these, the following items plus the innerbody shell and frames appear in the final weight statement classified as structural weight:

1. Bulkheads,
2. Access/Egress Provisions and Equipment Hatches,
3. Windows, and
4. Floors, shelves, etc.

Weights for the remaining structural items are associated with specific subsystem weights. For example, landing gear weight is included as a fraction of the Landing and Recovery Subsystem weight and, engine thrust structure, tanks (tip tanks,if applicable), tank support structure, etc. are all included in the Propulsion Subsystem weight. The same is true for docking and abort tower subsystem weight and is not broken-out in terms of specific structural component weights. However, a number of sizing models expressing structural weights have been developed for a majority of the above items. These models are obtained from semi-empirical equations based on statistical data compiled from existing hardware with some application of analytical equations. To further inform the reader on this procedure, the following paragraphs are devoted to the sizing models developed for windows and flooring.

6.13.4.1 Sizing Model for Windows - Structural weight attributed to windows consist of both window panes and support frames. Expressed as an equation, the window weight becomes:

$$W_{WF} = N_W \, [W_W + W_F]$$

where: W_{WF} = weight of windows and frames, pounds
N_W = number of windows, scalar
W_W = weight of window panes, pounds
W_F = weight of window frames, pounds

Further modification and development of this equation is accomplished by correlation to Mercury and Gemini window arrangement. The window concept utilized on these space vehicles consist of a three pane design. Panes of vycor glass are employed for the two outer panes and tempered glass of alumino silicate is considered for the inner pane. The entire arrangement is supported by a titanium frame work.

Using flat plate theory described by Reference 6.13-2, an analytical weight expression is obtained for rectangular window panes:

Where: $W_w = abt\rho$

W_w = pane weight, pounds

a = short dimensions of rectangular pane, inches

b = long dimensions of rectangular pane, inches

t = thickness of pane, inches, and

ρ = density of pane material, pounds per cubic inch

The thickness of the pane is expressed as a function of the allowable stress, F_{all}, and the maximum bending moment, M_{max}. This expression is developed as follows:

$$f_{max} = \frac{6M_{max}}{t^2}$$

$$t^2 = \frac{6M_{max}}{f_{max}}, \text{ and } t = \sqrt{\frac{6M_{max}}{f_{max}}};$$

$$M_{max} = \beta p a^2;$$

$$M.S. = \frac{F_{all}}{f} - 1 = 0, \; f = F_{all} ;$$

Therefore,

$$t = \sqrt{\frac{6\beta \; pa^2}{F_{all}}} = a\sqrt{\frac{p}{F_{all}}} \sqrt{6\beta}$$

Beta, (β) is the bending moment coefficient which is a function of plate dimensions, a and b, and the plate boundary conditions. Figure 6.13-5 illustrates this coefficient. Substituting the value for t into the weight equation, the following is obtained:

$$W_W = \rho b a^2 \sqrt{\frac{p}{F_{all}}} \sqrt{6\beta}$$

If this equation is rearranged as follows:

$$\frac{W_W}{\rho a^3} \sqrt{\frac{F_{all}}{p}} = \frac{b}{a} \sqrt{6\beta}$$

and plotted for a range of b/a ratios, a linear approximation is found for the left hand term or,

$$\frac{W_W}{\rho a^3} \sqrt{\frac{F_{all}}{p}} \cong \left(\frac{b}{a} - 0.45\right), \text{ for } .1 < \nu < .3$$

Figure 6.13-6 shows the linear relationship. Correlation to Gemini and Mercury windows, the following ratio is obtained for a three pane combination.

$$\rho\sqrt{\frac{p}{F_{all}}} = .00917$$

Making appropriate substitutions and re-arrangements, the final expression for weight of window panes becomes:

$$W_W = .00917 \; (ba^2 - .45a^3)$$

The weight attributed to window frames is correlated directly to Mercury hardware. This correlation is in terms of frame weight per unit of perimeter. The expression which results is:

FIGURE 6.13-5

MAXIMUM BENDING MOMENT
IN A RECTANGULAR PLATE

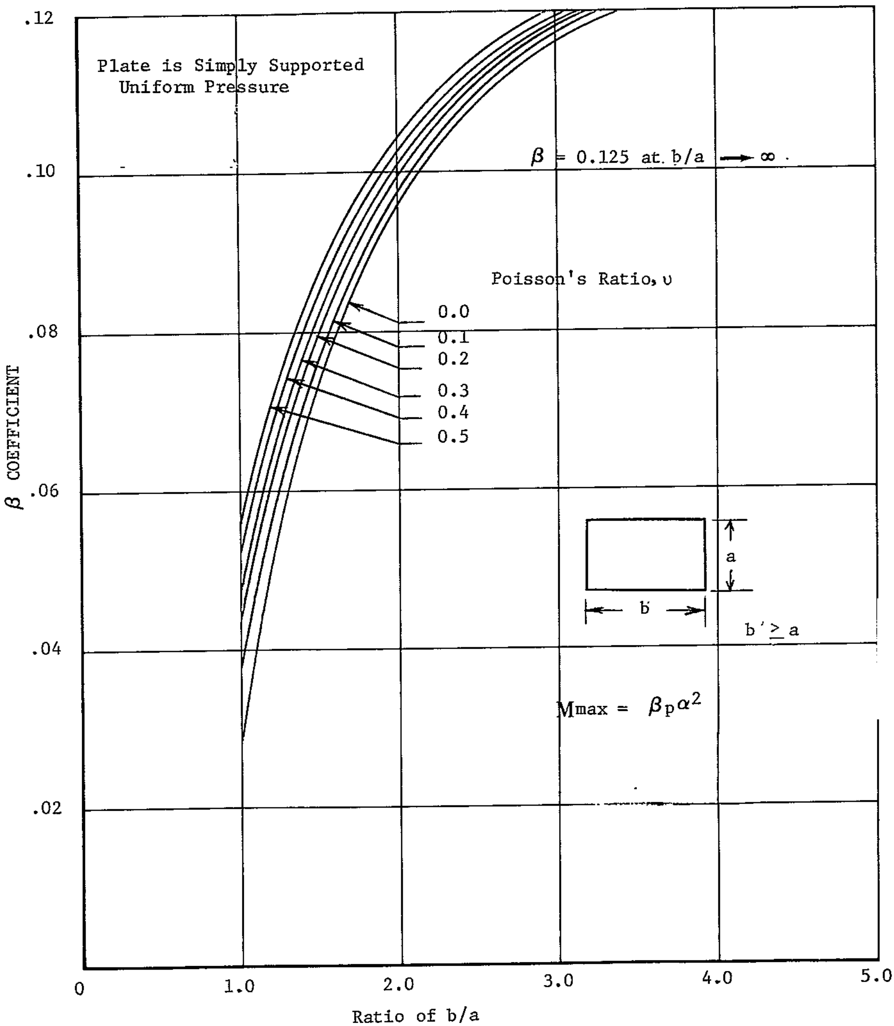
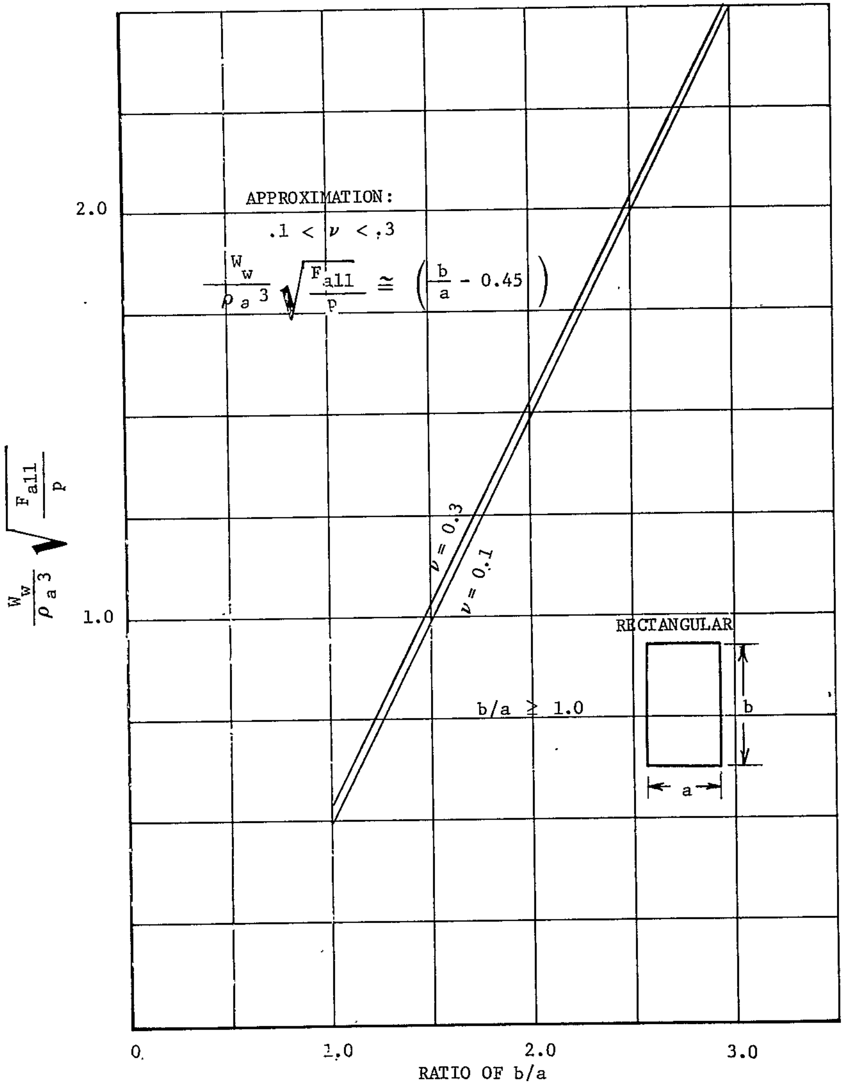


FIGURE 6.13-6

WINDOW WEIGHT APPROXIMATION



$$W_{F/Inch} = \frac{.068}{2} \left(b\right)^{.6}$$

The frame perimeter is expressed simply as:

$$d = 2\ (a + b).$$

Combining these two expressions results in the weight contribution for each window frame:

$$W_{WF} = .068\ (a + b) \left(b\right)^{.6}$$

Having derived the weight expressions attributed to panes and frames the final equation for window weight becomes:

$$W_{WF} = N_W\ [.00917\ (ba^2 - .45a^3) + .068\ (a + b)\ b^{.6}]$$

6.13.4.2 Sizing Model for Flooring - The structural weight attributed to flooring consists of basic flooring structure, sidewalls, shelves, etc. The equation defining these weights is a constant 1.5 pounds per square foot times the associated floor area:

$$W_{FL} = 1.5\ (K_F)\ (S).$$

where:

W_{FL} = weight of flooring, shelves, wall, etc., pounds

K_F = ratio of flooring area to spacecraft platform area, scalar, and

S = spacecraft platform area, square feet.

6.13.5 Flow Diagrams - The following figures 6.13-7 and 6.13-8 present logic diagrams for sizing the shell and the frame in the two supporting subroutines to the structural subroutine.

6.14 PROPULSION ROUTINE - The propulsion routine calculates the thrust levels required, the propellant weight, the propellant tank volumes, and the weights of the propulsion system components necessary to perform the selected spacecraft maneuvers. The subroutine contains the calculation procedures for several propulsion systems. Table 6.14-1 shows the systems that are available and the functions they are capable of:

The following is a brief description of the various maneuvers.

Boost - Provides all or part of the ΔV (incremental velocity) needed for initial orbit insertion.

FLOW DIAGRAM FOR SHELL SIZING MODEL

FIGURE 6.13-7

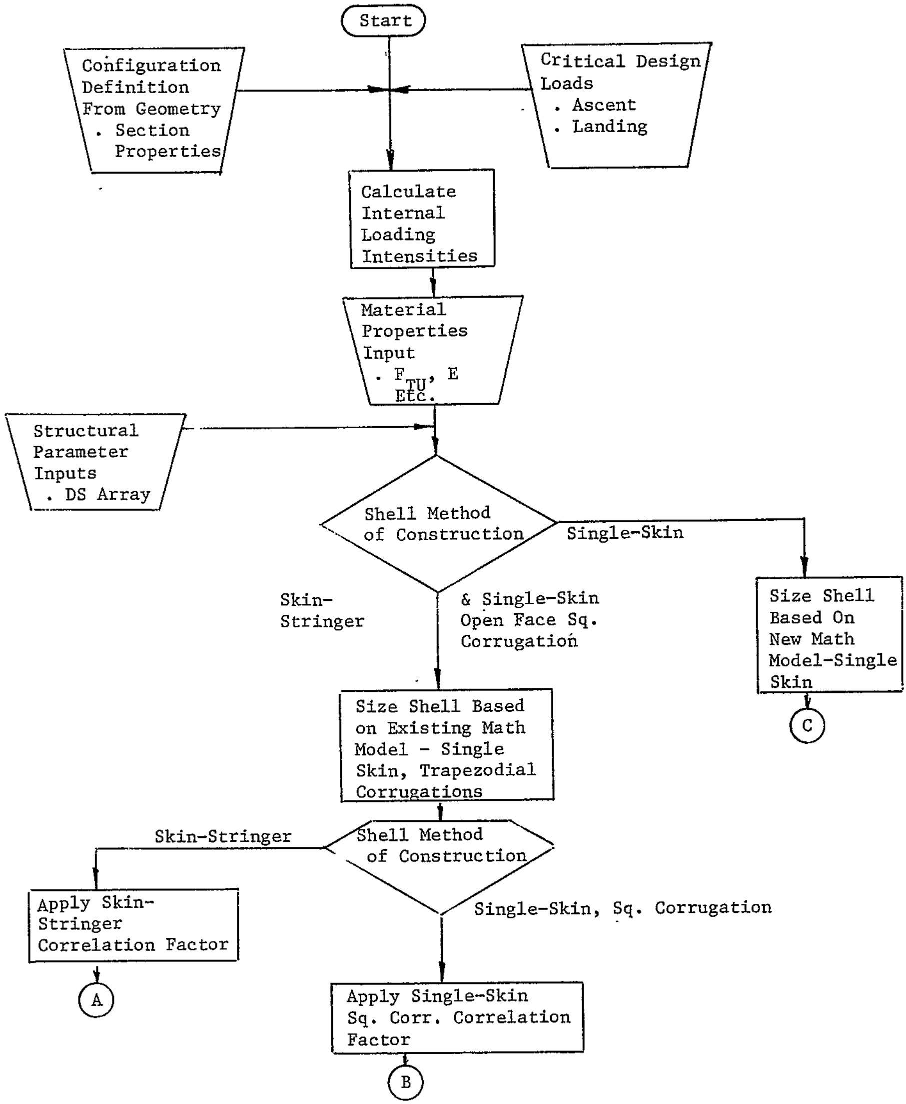


FIGURE 6.13-7 (Contd)

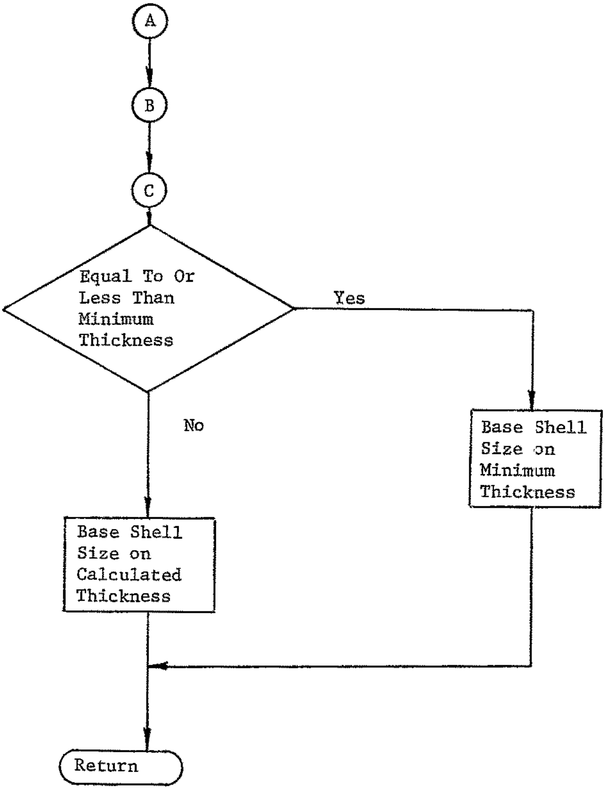


FIGURE 6.13-8

FLOW DIAGRAM FOR FRAME SIZING MODEL

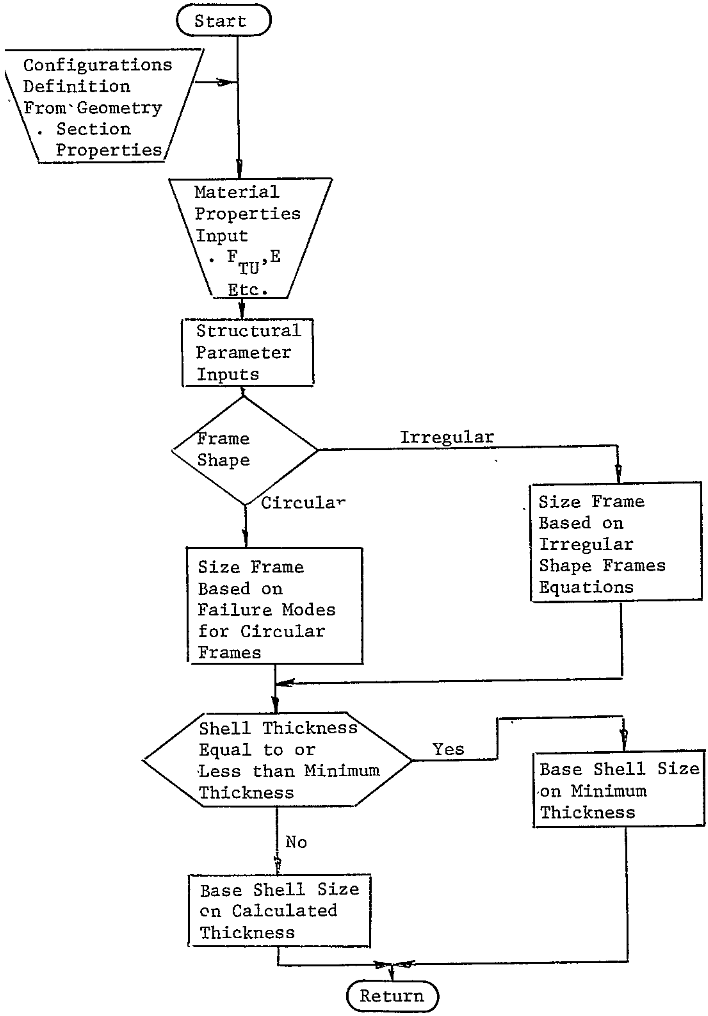


TABLE 6.14-1

SYSTEMS AND FUNCTIONS

Systems \ Functions	Boost	Launch Escape	Ascent	Docking	Orbit Attitude Cont.	Phasing	Retro	Entry Attitude Cont.	Landing Assist
Integral Boost	X								
Tip Tank	X								
Launch Escape		X							
Main Maneuver			X			X	X		
Orbit Attitude Control					X			X	
Vernier Maneuver			X	X	X	X	X		
Liquid Retro						X	X		
Solid Retro						X	X		
Entry Attitude Control								X	
Landing Assist									X

Launch Escape - Provides a means of crew escape in the advent of booster malfunction.

Ascent - Provides ΔV necessary to ascend or maneuver from injected orbit prior to docking.

Docking - Provides 3 axis translation to dock with space station.

Orbit Attitude Control - Provides 3 axis rotation to maintain attitude during orbit.

Phasing - Provides ΔV necessary to change orbital plane or period for landing purposes.

Retro - Provides ΔV necessary to de-orbit.

Entry Attitude Control - Provides 3 axis rotation to maintain attitude during re-entry.

Landing Assist - Provides vertical velocity attenuation prior to touchdown for ballistic vehicles. Provides glide range extension for lifting bodies.

To use the propulsion routine, the spacecraft maneuvers must first be defined. Propulsion systems are then selected to perform the desired functions and program inputs such as ΔV, F/W, propellant type, etc. are provided. The propulsion routine then sizes and weighs the selected systems.

The propellant quantity and thrust level of each system are calculated from the performance parameters input, and are used to calculate the weights of the components which comprise the system. The weight of the propulsion system components is calculated either by empirically derived equations based on existing hardware or vendor supplied parametric data, or by equations theoretically derived based on fundamental relationships. In the following paragraphs, the weight calculations procedures used in the propulsion routine are defined and the functional description and calculation logic flow for each system is discussed.

6.14.1 Calculation Procedure

6.14.1.1 System Performance Parameters - Two basic calculations are performed in each of the propulsion systems. These are the propellant quantity and the thrust level required. These two values are used directly or indirectly as inputs to enable calculation of the component weights.

Propellant Quantity (Weight and Volume) - The propellant weight is calculated as a function of ΔV (except for attitude control), landing assist, and launch escape functions) and I_{sp} (specific impulse), and is based on the vehicle weight which is determined in the mass properties routines. Depending on whether the mass properties routine is tabulating the weight prior to the maneuver or after it, one of the following equations is used:

$$Wp = Wo\ (1 - 1/B) \text{ or } W_p = Wbo\ (B - 1)$$

where:
W_p = propellant weight

Wo = initial weight

Wbo = final weight

$B = e^{\Delta V/g\ Isp}$

For those systems which perform an attitude control, landing assist, or launch escape function, the propellant weight is calculated from total impulse (which is input as a percentage of vehicle weight) and I_{sp},

$$Wp = It/Isp$$

Propellant volumes are calculated for the liquid propellant systems only. They are used to determine the propellant tank volumes required, which are needed to cost the tanks. Solid propellant systems are not costed on a volume basis, so the propellant volume is not required.

Propellant volumes for the liquid propellant systems are calculated by:

$$V_{ox} = wp/Pox\ (1 + 1/R)$$

$$Vf = wp/Pf\ (1 + R)$$

where:

Vox = oxidizer volume $\qquad$ Pox = oxidizer density

Vf = fuel volume $\qquad$ P_f = fuel density

R = mixture ratio

The liquid propellant characteristics are stored as the PROPEL Matrix The solid propellant performance characteristics are not in the PROPEL Matrix, but are input directly into the program.

Thrust - The thrust levels for all systems are either calculated or input as a specific value. If the thrust level is to be calculated, it is calculated on the basis of an input thrust to weight ratio for all systems except those performing an attitude control function. The thrust levels for attitude control functions are calculated from an input angular accelerations and moment arm by:

$$F = (I)\ (\alpha)\ (L)\ (\#\ \text{engines})$$

where: F = thrust per engine
I = moment of inertia
α = angular acceleration
L = moment arm

The moment of inertia is calculated in the mass properties routine. Engine configuration and number of engines are stored in the ENGCON matrix.

6.14.1.2 Liquid Systems - The liquid propulsion systems are comprised of the following basic components:

Engines
Thrust Structure
Gimbal system
Propellant Tanks
Pressurization System
Bulkheads
Support and Installation
Miscellaneous (lines, valves, etc.)

Engines - The engines provide the thrust required to perform the various spacecraft maneuvers.

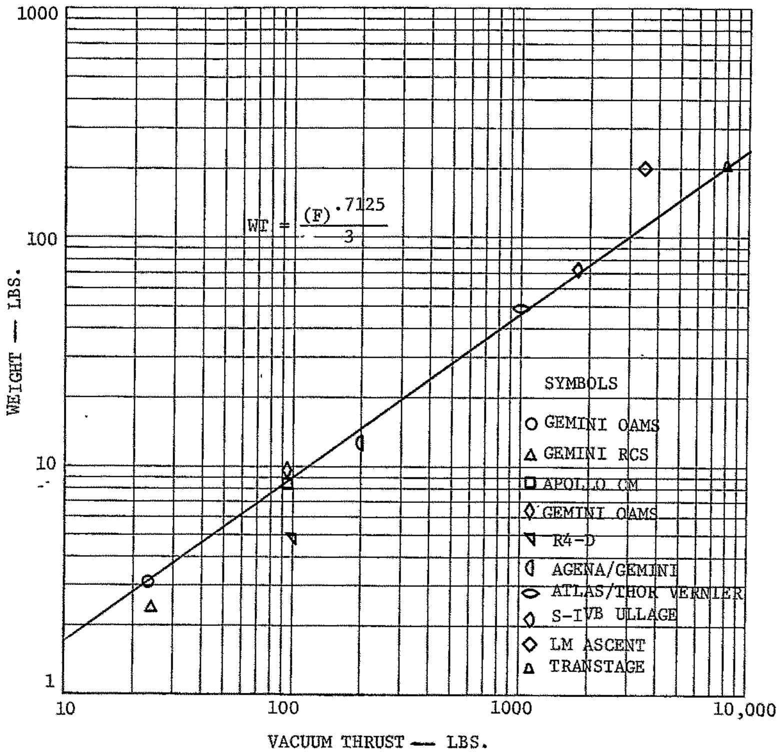
All engine weights, except those used in the integral boost system, are calculated by the following equation. Figure 6.14-1 shows the data utilized in deriving the equation:

$$\text{Engine weight} = \frac{(F)^{.7125}}{3}$$

The weights for the engines used in the integral boost system are calculated by one of the following expressions:

FIGURE 6.14-1

ATTITUDE CONTROL AND MANEUVER ENGINE WEIGHTS



(a) For advanced high Pc bell nozzle engines using O_2/H_2, F_2/H_2, or $FLOX/CH_4$,

Engine weight = 580 + (.008677) (F)

(b) For advanced bell nozzle engines using NTO/A-50,
Engine weight = $(.000883)\ (F)^{1.2}$

(c) For Aerospike engines using O_2/H_2,
Engine weight = $[(20.25)\ (D) - 20]\ (F/10^5)^n$

Where: $D = 188 + 25 \left(\frac{F}{10^6}\right)^{.46}$

$n = 10.38/D^{.734}$

F = Vacuum thrust/engine

The above expressions were derived from vendor supplied parametric data, for high area ratio (60-100) upper stage engines and are different from the empirical equation used for the other systems because of the large thrust levels and the unique design of these engines.

Thrust Structure - The thrust structure receives the thrust produced by the engine and transmits it to the structure of the vehicle. It also serves as the mounting for the engine. The thrust structure weight for all systems is calculated from empirical equations. Except for the integral boost system the thrust structure weights are calculated by:

$$\text{Thrust structure weight} = (.308)\ (\#\ \text{engines})\ (F)^{.55}$$

Figure 6.14-2 shows the data utilized in deriving this equation.

The thrust structure weight for the high thrust level integral boost system is calculated by:

$$\text{Thrust Structure weight} = (.00000773)\ (F)^{1.47}$$

where: F = Total vacuum thrust

Figure 6.14-3 shows the data utilized in deriving this equation.

Gimbal System - The gimbal system provides the necessary mechanism to gimbal the engine and thus provide a controlling force to the vehicle.

The gimbal system weight is calculated by the following empirical equation.

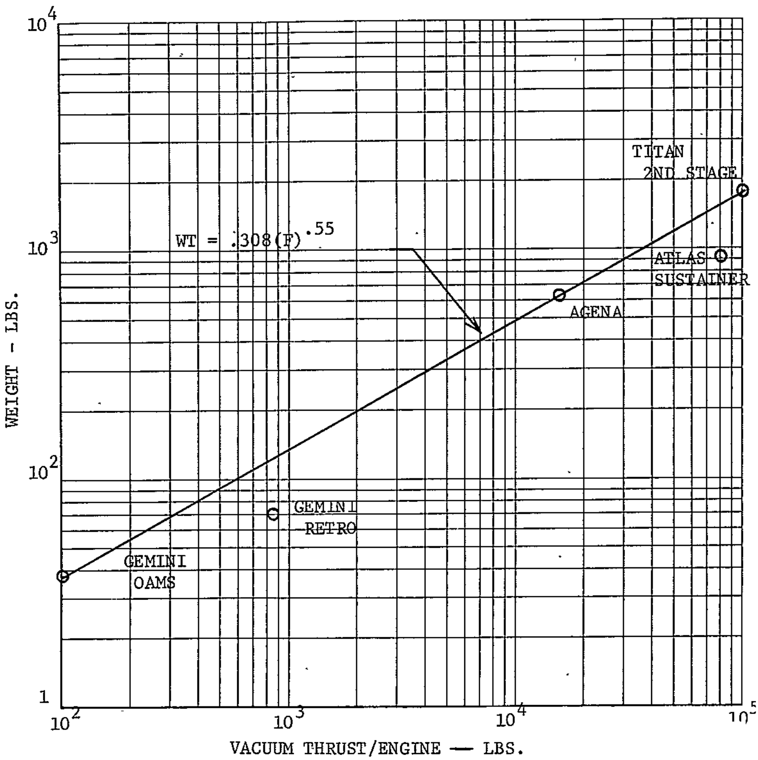
Gimbal System Weight = (.15) (Total engine weight)

Figure 6.14-4 shows the data utilized to obtain typical hydraulic actuator system weights, to which an additional assumed 5% of total engine weight was added to account for engine mountings for the gimbal system.

Propellant Tanks - The propellant tank weights are calculated by the following equations, with the larger value being used in the spacecraft weight derivation.

FIGURE 6.14-2

THRUST STRUCTURE WEIGHT



Propellant tank weight = (.00788) $\left(\frac{Wp}{P_b}\right)$ (P) (shape factor)

where: Wp = propellant weight

P_b = propellant bulk density

P = Tank pressure

FIGURE 6.14-3

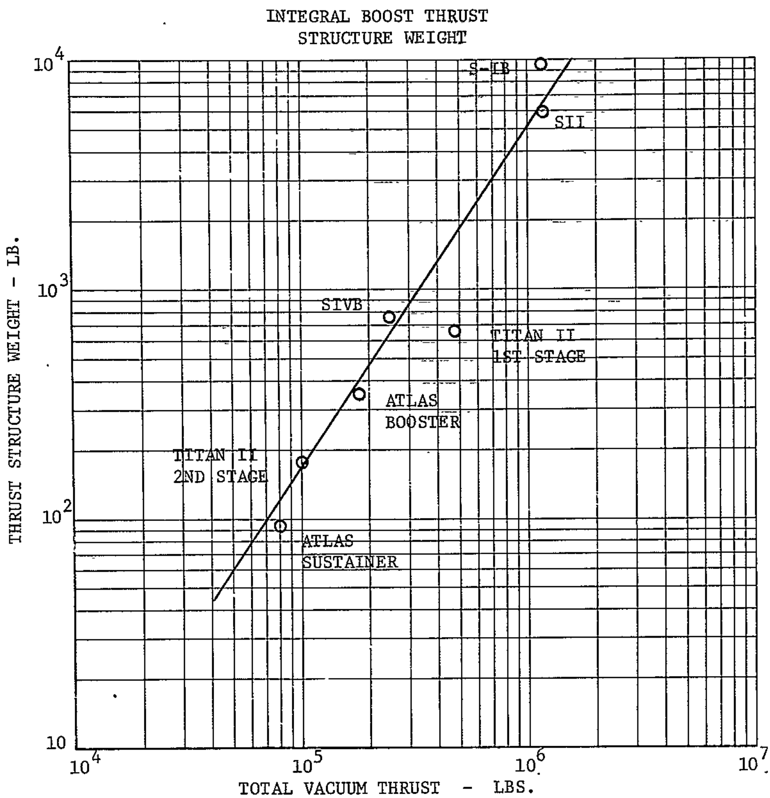
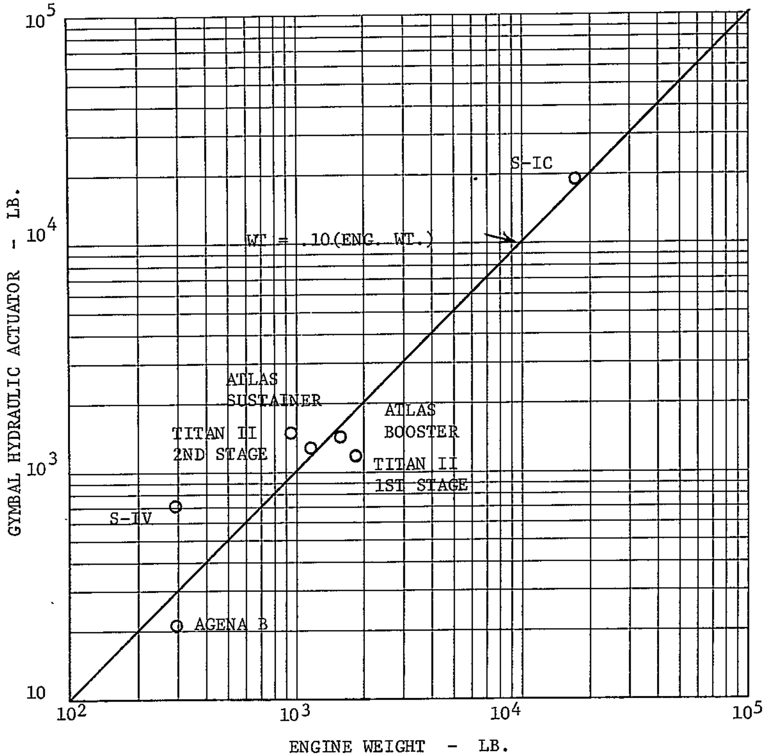

INTEGRAL BOOST THRUST
STRUCTURE WEIGHT
THRUST STRUCTURE WEIGHT - LB.
TOTAL VACUUM THRUST - LBS.
10
10^2
10^3
10^4
10^5
10^6
10^7
S-IB
SII
SIVB
TITAN II
1ST STAGE
ATLAS
BOOSTER
TITAN II
2ND STAGE
ATLAS
SUSTAINER

FIGURE 6.14-4

GIMBAL HYDRAULIC ACTUATOR SYSTEM WEIGHTS



The above is a theoretical equation based on the ideal weight of a sphere (shape factor = 1) using a non-optimum factor of 1.35. For a tapered skin torroidal tank the shape factor, based on theory, is 1.3.

$$\text{Propellant tank weight} = (1.21)(\#\ \text{tanks})^{1/3} \left(\frac{W_p}{P_b}\right)^{2/3} (\text{shape factor})$$

The above equation is based on a minimum gage tank and is included in the program to insure realistic weights for small and/or low pressure systems.

Pressurization System - The pressurization system is used to provide the net positive suction head for pump fed engines or the feed pressure required for pressure fed engines.

All pressurization systems, except the integral boost system, are calculated by the sum of gas weight, bottle weight, and miscellaneous weight.

(a) $$\text{Gas weight} = \frac{(\gamma)(P)\ (W_p)\ (3000)}{(T)\ (R)\ (P_B)\ (3000-P)}$$

where:
- γ = pressurant specific heat ratio
- P = propellant tank operating pressure
- W_p = total propellant weight
- T = initial pressurant temperature
- R = gas constant
- P_B = propellant bulk density

The above is Ring's theoretical equation for the gas weight needed to pressurize a given volume to a given pressure starting at a pressure of 3000 psia.

(b) $$\text{Bottle weight} = \frac{(23.3)\ (\gamma)\ (P)\ (W_p)}{(P_B)\ (3000-P)}$$

The above is a theoretical equation based on the ideal weight of a spherical bottle and a non-optimum factor of 1.35.

(c) Miscellaneous = 15 pounds This is an arbitrary 15 pounds to account for miscellaneous plumbing and controls in the pressurization system.

For the integral boost system, the following empirical equation is used to calculate the weight of the pressurization system.

Pressurization System Weight = (C) (W_p)

where:

W_p = Propellant weight

C = Pressurization system factor

For O_2/H_2, C = .005 (Based on S-IVB type pressurization system)

For NTO/A-50, C = .003 (Based on Titan II type pressurization System)

The pressurization system weight of the integral boost system is different from the ambient stored gas systems due to the weight advantages of utilizing cryogenic gas storage temperatures and/or heated vapors for the pressurant and autogeneous gas generator pressurants for large cryogenic or storable systems respectively.

Bulkhead - Bulkheads are used in those cases where an integral propellant tank is desired. In an integral tank design, the bulkheads are used to seal off sections of the structure, with the sealed off section being used for a propellant tank. Bulkhead weights are calculated by the following equations with the larger value being used in the spacecraft weight derivation.

(a) Bulkhead weight = 4.71 R^2

where R = Bulkhead radius

The above equation represents a minimum gage bulkhead.

(b) Bulkhead weight = (.0153) (P) $(R)^3$

where P = Tank operating pressure

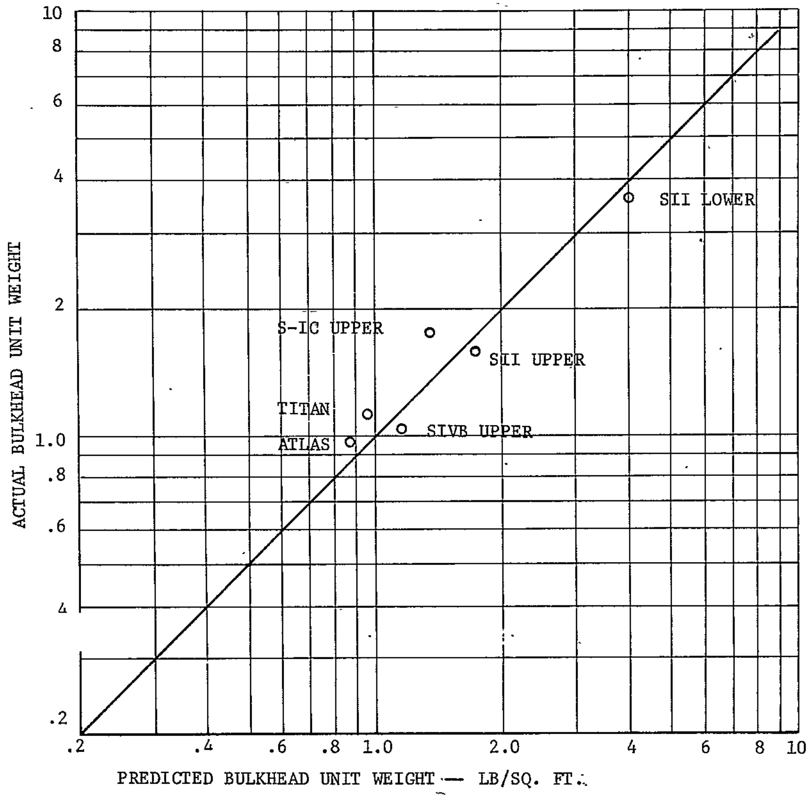
R = Bulkhead radius

The above equation is theoretical and figure 6.14-5 shows the agreement of the above equation to existing hardware.

Support and Installation - The support and installation weight is the weight of the brackets and mountings which support the propellant tanks. The weight is calculated by the following empirical equation:

IGURE 6.14-5

BULKHEAD WEIGHT AGREEMENT



Support and Installation Weight = $(.931)\ (W_p)^{.65}$

Figure 6.14-6 shows the curve fit of the support and installation weight vs. the total supported weight. A propellant mass fraction of .9 was assumed for typical propellant tankage, and was substituted into the curve fit expression to yield the final equation.

Miscellaneous - This weight is to account for the propulsion system plumbing (i.e. lines, valves, etc.). The miscellaneous weight for all systems, except the integral boost system, is calculated by the following empirical equation:

Miscellaneous weight = $.50(F)^{.458}(N)$

where: F = Thrust/Engine

N = Number of Engines

Figure 6.14-7 shows the existing hardware data utilized in obtaining the equation.

For the integral boost system, the miscellaneous weight is calculated by the following empirical equation:

Miscellaneous weight = $.0214\ (F)^{.888}$

where: F - Total thrust

Figure 6.14-8 shows the existing hardware data utilized in obtaining this equation.

6.14.1.3 Solid Propellant Systems - The solid propellant systems are;

Solid Retro system
Landing Assist system
Launch Escape system

In all of these systems, the total system weight is comprised of a loaded motor weight and a thrust structure.

Loaded Motor - The loaded motor represents the propellant weight, the case weight, and the ignitor weight. The weight is computed from:

Loaded motor weight = $\frac{W_p}{m.f.}$

FIGURE 6.14-6

GEMINI SYSTEMS SUPPORT STRUCTURE WEIGHTS

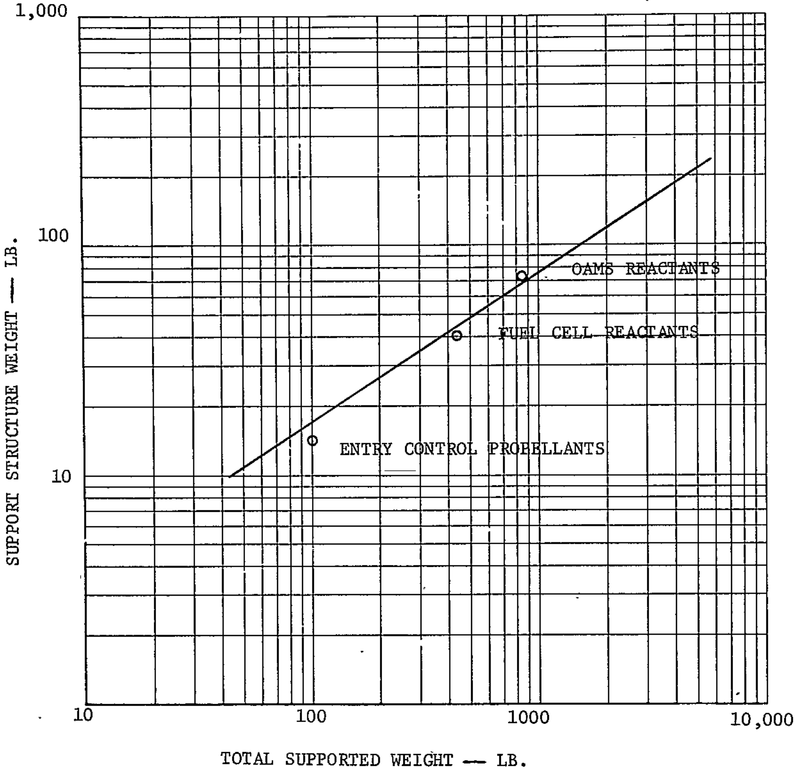


FIGURE 6.14-7

MANEUVER PROPULSION ENGINE WEIGHT

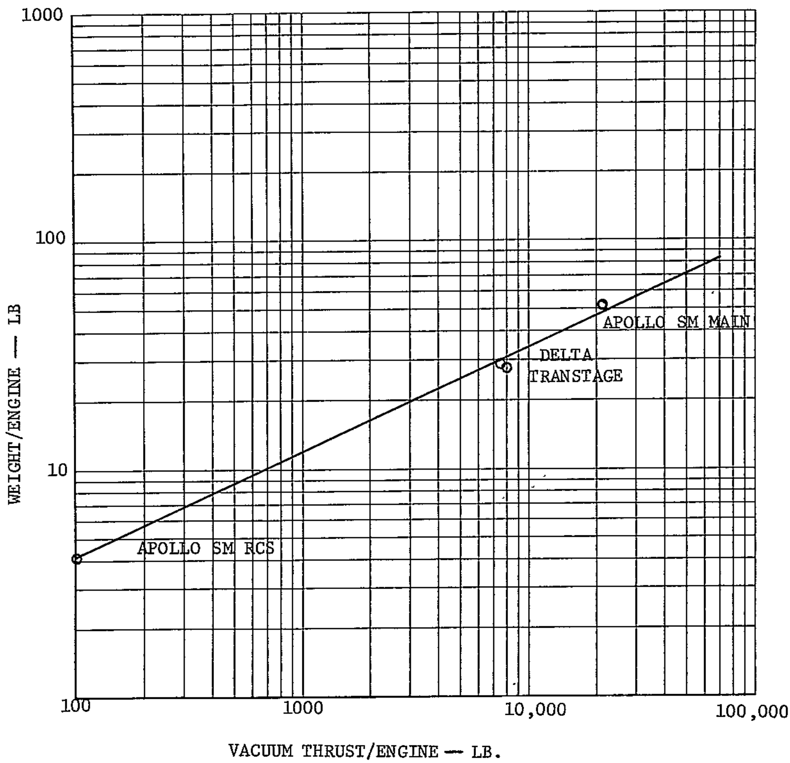
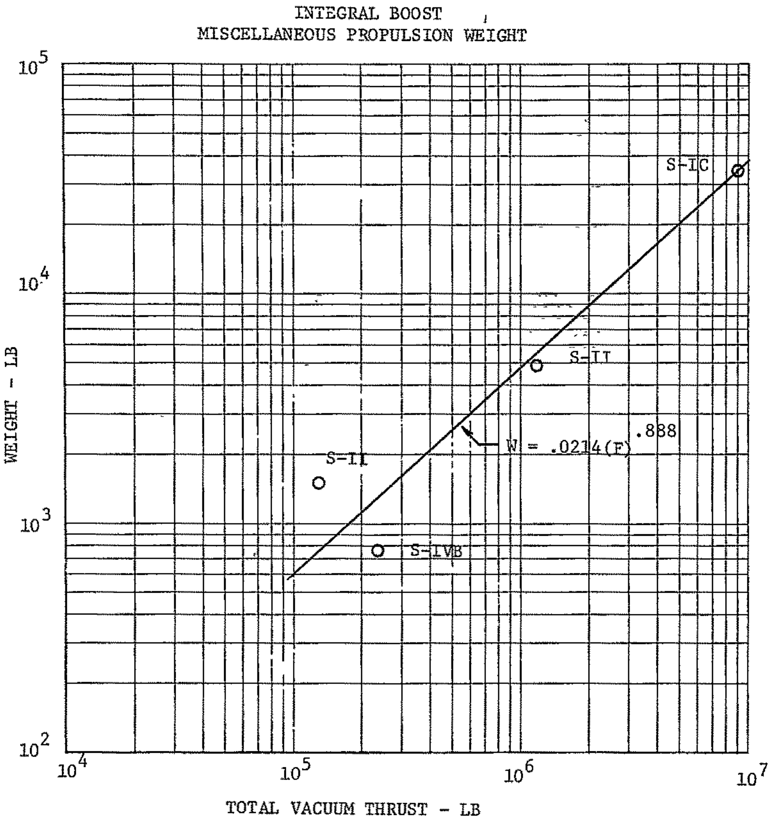


FIGURE 6.14-8


INTEGRAL BOOST
MISCELLANEOUS PROPULSION WEIGHT
WEIGHT - LB
TOTAL VACUUM THRUST - LB
10^2
10^3
10^4
10^5
10^4
10^5
10^6
10^7
S-IC
S-II
S-II
S-IVB
$W = .0214(F)^{.888}$

where: W_p = Propellant weight

m.f. = Propellant mass fraction

Thrust Structure - The thrust structure weight is calculated by the same equation as the liquid systems (see Figure 6.14-2)except for the externally mounted low altitude launch escape system. These thrust structure weights are calculated by the following empirical equations:

$$\text{Tower Thrust Structure Weight} = (.029)\ (F)^{.88}$$

$$\text{Strap-On Thrust Structure Weight} = (627)\ \left(\frac{F}{10^6}\right)^{.6}$$

where: F = Total thrust

6.14.2 Program Logic - This section contains a logic diagram for each system and briefly describes pertinent facts concerning the calculation methodology of each.

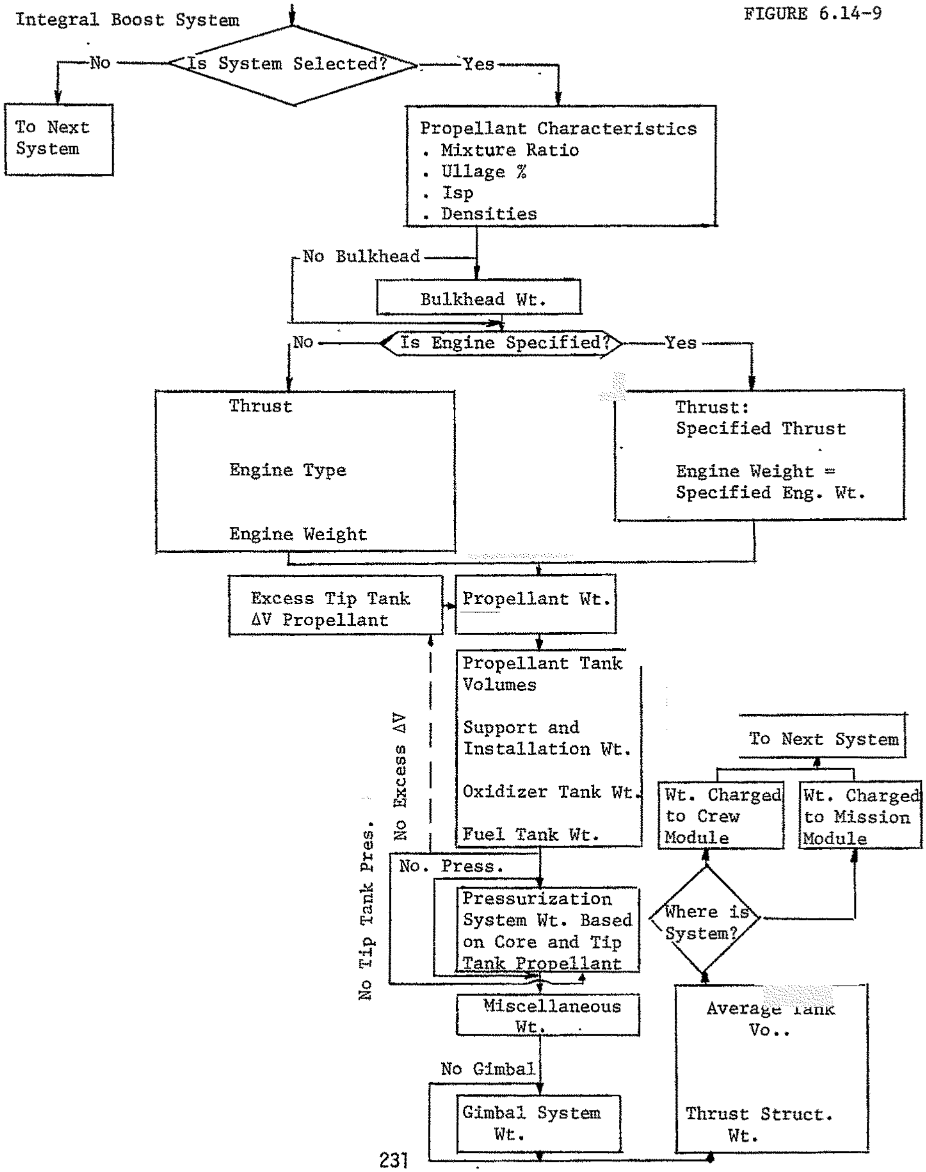
In operating the program, any of the systems may be bypassed if they are not desired. All systems can be located in either the crew module or the mission module with the total system weight charged accordingly unless otherwise noted.

6.14.2.1 Integral Boost System - The integral boost system is used to provide the boost maneuver. Boost system components are a significant portion of the spacecraft weight because of the large size of boost systems. Therefore, boost system weight differences due to choice of engine, propellant type, and tank geometry, must be accounted for. This portion of the propulsion routine has these unique characteristics:

1. There is a choice of engine weight equations depending on what type of engine and propellant is desired.
2. The fuel and oxidizer tanks are calculated separately and may be of different configurations.

Figure 6.14-9 shows the logic of the integral boost system. The pressurization system for the integral boost system is based on the propellant weight of both the integral boost system and the total tip tank system, if the latter is incorporated int the vehicle configuration.

FIGURE 6.14-9


Integral Boost System
Is System Selected?
No
Yes
To Next System
Propellant Characteristics
. Mixture Ratio
. Ullage %
. Isp
. Densities
No Bulkhead
Bulkhead Wt.
Is Engine Specified?
No
Yes
Thrust
Engine Type
Engine Weight
Thrust:
Specified Thrust
Engine Weight =
Specified Eng. Wt.
Excess Tip Tank
ΔV Propellant
Propellant Wt.
Propellant Tank Volumes
Support and Installation Wt.
Oxidizer Tank Wt.
Fuel Tank Wt.
No Excess ΔV
No Tip Tank Pres.
No. Press.
Pressurization System Wt. Based on Core and Tip Tank Propellant
Miscellaneous Wt.
No Gimbal
Gimbal System Wt.
To Next System
Wt. Charged to Crew Module
Wt. Charged to Mission Module
Where is System?
Thrust Struct. Wt.

The integral boost system may be used by itself, but the tip tank system, described below, can be used only with the integral boost system. Since both systems use the same engines, the same propellant must be selected for both systems. If the integral boost system is selected to be in the crew module, there can be no mission module.

6.14.2.2 Tip Tank System - The tip tank system provides externally mounted tankage to provide additional propellant to the integral boost system. The tip tanks are jettisoned after they are used.

Figure 6.14-10 shows the logic of the tip tank system. The main portion of the logic is taken up with sizing the tip tank. Provisions exist to constrain the length, diameter or L/D of the tip tank. Depending on the constraints, the tip tanks may not be able to accommodate enough propellant to supply the ΔV desired. In this case, the additional ΔV is added to the integral boost system or can be neglected. If this ΔV is neglected, the ground launched boost stage must be resized to account for the difference.

The tip tanks are considered to be cylinder with a 15° half angle cone on top, and a hemispherical end cap on the bottom. The volume of this shape is:

$$V = \frac{d^3 \ (L/d - 1.4106)\Pi}{4}$$

where: d = tip tank diameter

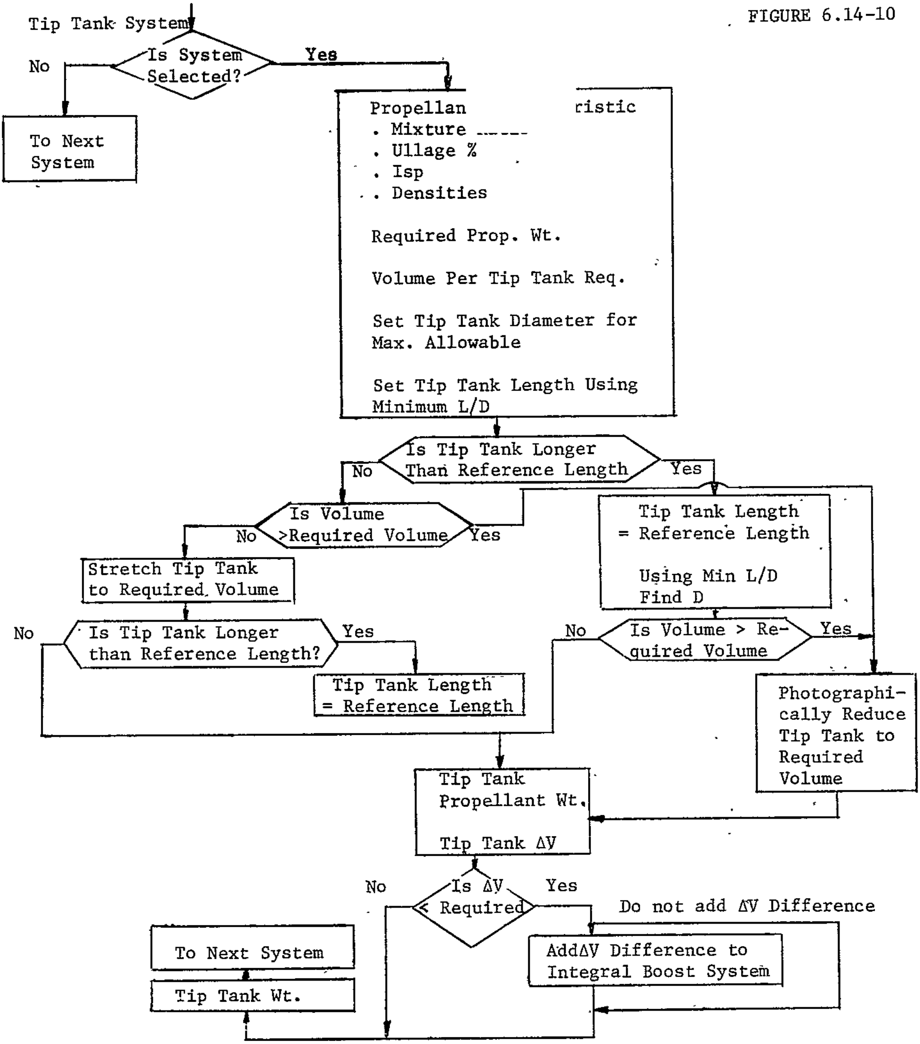
L = tip tank length (measured from tip of cone to bottom of hemisphere)

The minimum L/d that can be used for the tip tanks is 2.37. This ratio will yield a 15° cone and a hemisphere with no cylindrical section between them. To reduce the L/d from 2.37 is meaningless for this configuration.

6.14.2.3 Launch Escape System - The launch escape system provides the function of crew escape in the advent of boost malfunction. There are two modes of launch escape: low altitude and high altitude.

The low altitude system may be installed in any of 3 ways; tower mounted, strap-ons, or internally mounted. The tower and strap-on systems are jettisoned at high altitude. The mass properties routine contains a jettison factor to account for the jettisoned weight.

FIGURE 6.14-10



The high altitude system is only internally mounted. For modular vehicles, the retro system can be utilized as the high altitude launch escape system. Figure 6.14-11 shows the logic of the launch escape system.

6.14.2.4 Main Maneuver System - The main maneuver system is capable of providing any or all of the following maneuvers: ascent, phasing, and retro. Figure 6.14-12 shows the logic of the main maneuver system. The thrust level is determined by whichever maneuver requires the most thrust. The system has the option of either integral or non-integral propellant tanks, but both fuel and oxidizer tanks must be of the same shape.

6.14.2.5 Orbit Attitude Control System - The orbit attitude control system is capable of providing orbital attitude control and/or entry attitude control. Figure 6.14-13 shows the logic of the orbit attitude control system. If the thrust is to be calculated instead of specified, the moment arm used in the calculations may be specified, or can be input as a percent of vehicle length. The thrust level used is the largest required of the two attitude control requirements.

6.14.2.6 Vernier Manuever System - The vernier maneuver system is capable of performing all, or any combination of the following maneuvers: ascent, docking, retro, orbital attitude control, and phasing. Figure 6.14-14 shows the logic of the vernier maneuver system. The thrust level used is the largest required for the selected maneuvers.

6.14.2.7 Liquid Retro System - The liquid retro system can provide the phasing and/or retro function. In operation, it is similar to the main maneuver system. Figure 6.14-15 shows the logic of the liquid retro system. The system may be in the crew module, mission module, or a retro adapter with the total system weight charged accordingly.

6.14.2.8 Solid Retro System - The solid retro system can provide the phasing and/or retro function. Its usage is similar to the liquid retro system. Figure 6.14-16 shows the logic of the solid retro system. The system may be in the crew module, mission module, or a retro adapter with the total system weight charged accordingly.

6.14.2.9 Entry Attitude Control System - The entry attitude control system performs the function of entry attitude control. In operation, it closely

FIGURE 6.14-11

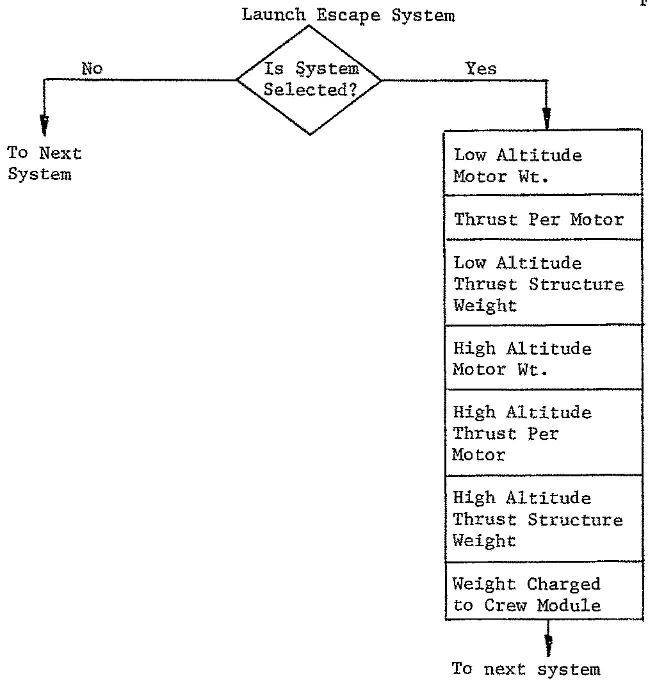
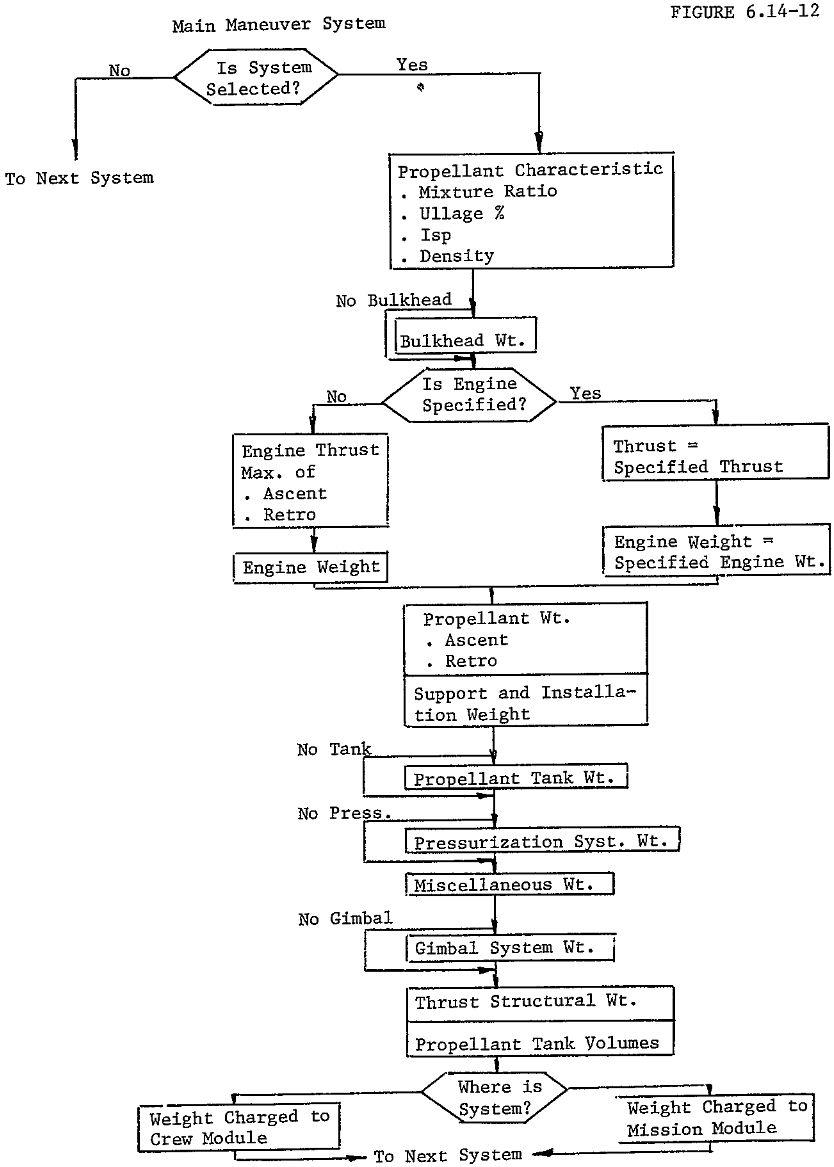


FIGURE 6.14-12



Orbit Attitude Control System

FIGURE 6.14-13

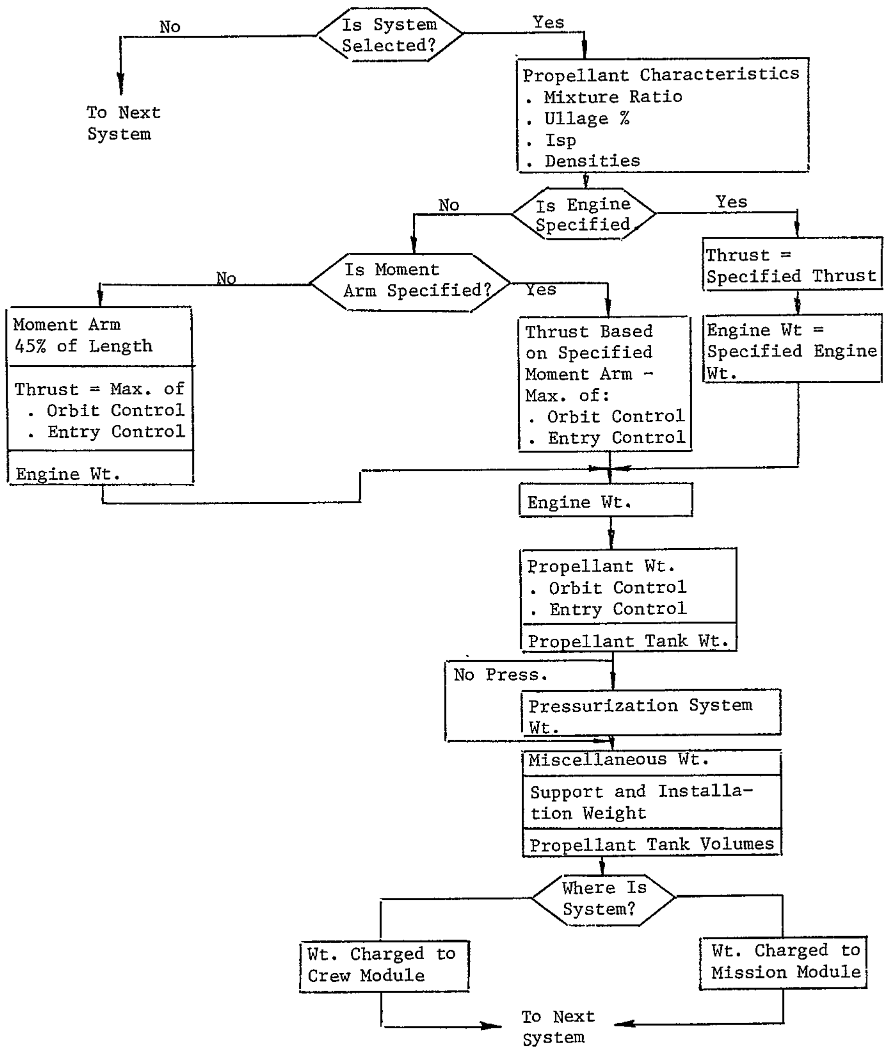


FIGURE 6.14-14

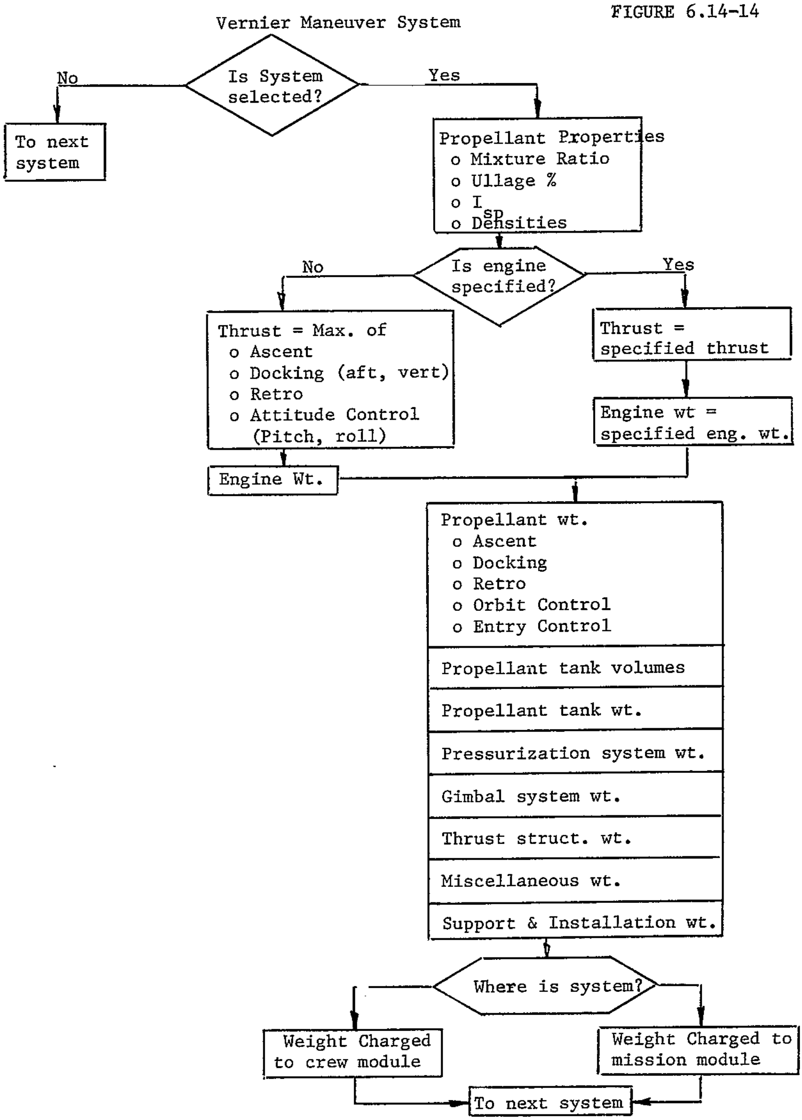


FIGURE 6.14-15

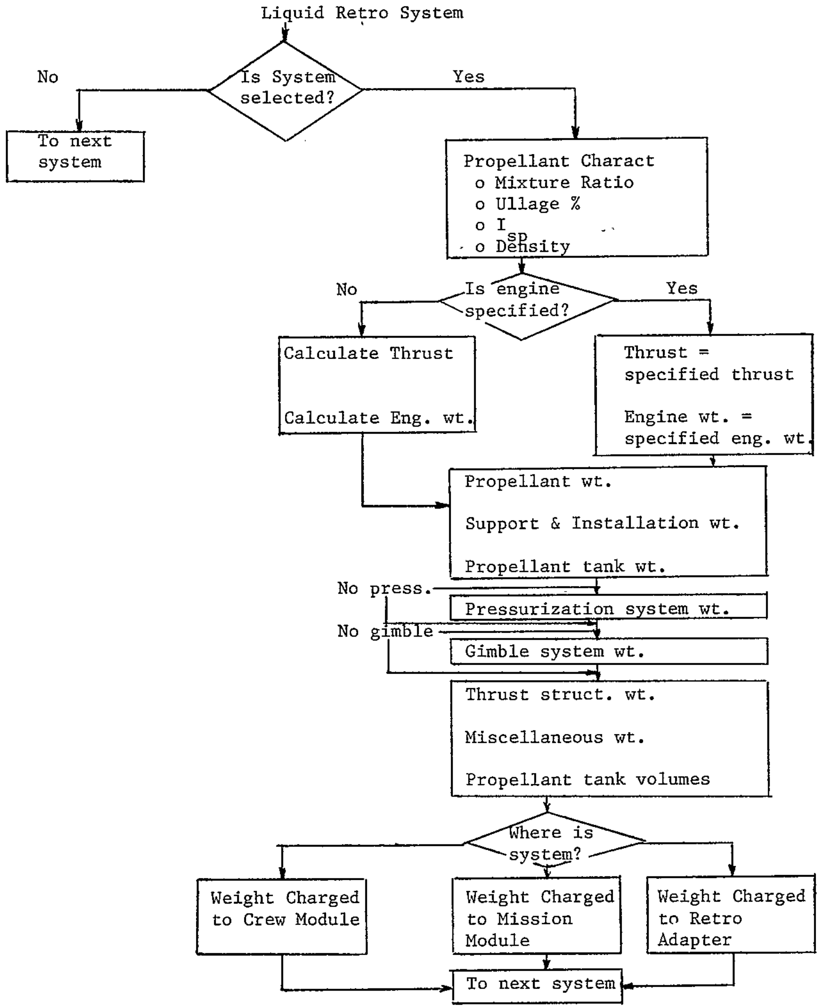
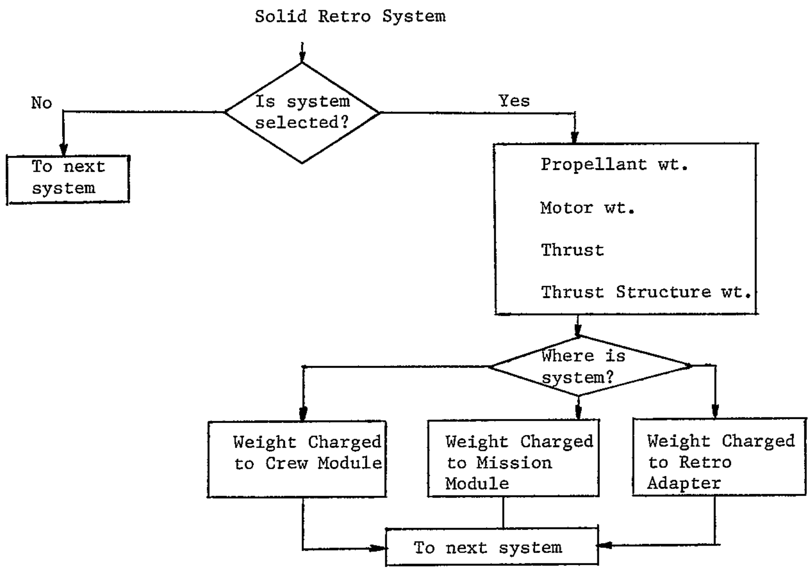


FIGURE 6.14-16



resembles the orbit attitude control system. Figure 6.14-17 shows the logic of the entry attitude control system. If engine thrust is to be calculated, the moment arm may be specified or input as a percent of vehicle length. The system is in the crew module and may be made redundant, in which case, twice the total system weight is charged to the crew module.

6.14.2.10 Landing Assist System - The landing assist system serves a different function depending on whether the vehicle is a ballistic or a lifting body. For a ballistic vehicle, the landing assist system attenuates the vehicles vertical velocity just prior to touchdown. For a lifting body, the landing assist system provides thrust for powered flight to increase the glide range of the vehicle.

The type of function provided by the landing assist system must be selected. For ballistic vehicles, vertical velocity attenuation should be selected and for lifting bodies, glide range extensions is used. Figure 6.14-18 shows the logic of the landing assist system.

If the system is selected for vertical velocity attenuation, the vertical velocity is calculated by:

$$\text{Vertical velocity} = 24 \sqrt{w/6700}$$

where w = landing weight

The above equation is based upon the characteristics of the Apollo parachute configuration. The landing weight must be larger than 6700 lbs or the logic will bypass this calculation. The It/w ratio used to calculate the propellant weight is found from:

$$It/w = (F)\ (V)$$

where F = The impulse factor
V = vertical velocity

If the system is selected for glide range extension, the It/w ratio used to calculate the propellant weight is calculated by:

$$It/w = (F/W)\ (t)$$

where F/w = the desired thrust to weight ratio
t = burn time

The total system weight is charged to the crew module for both types of systems.

FIGURE 6.14-17

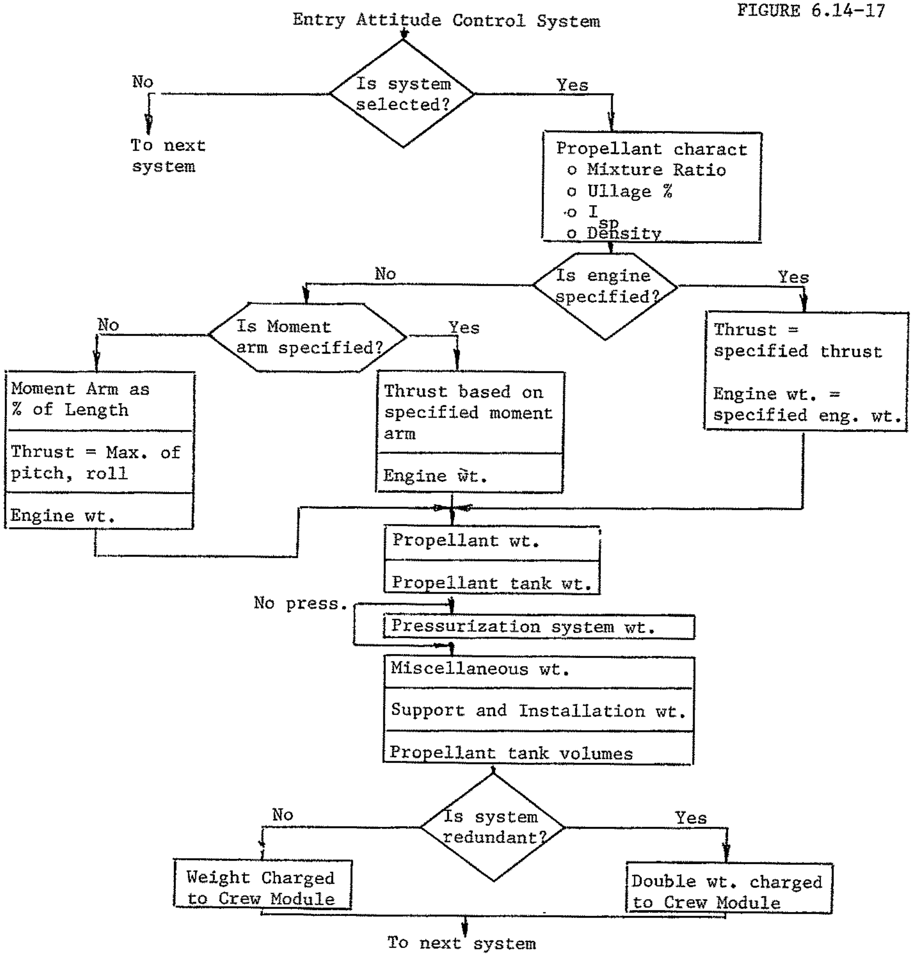
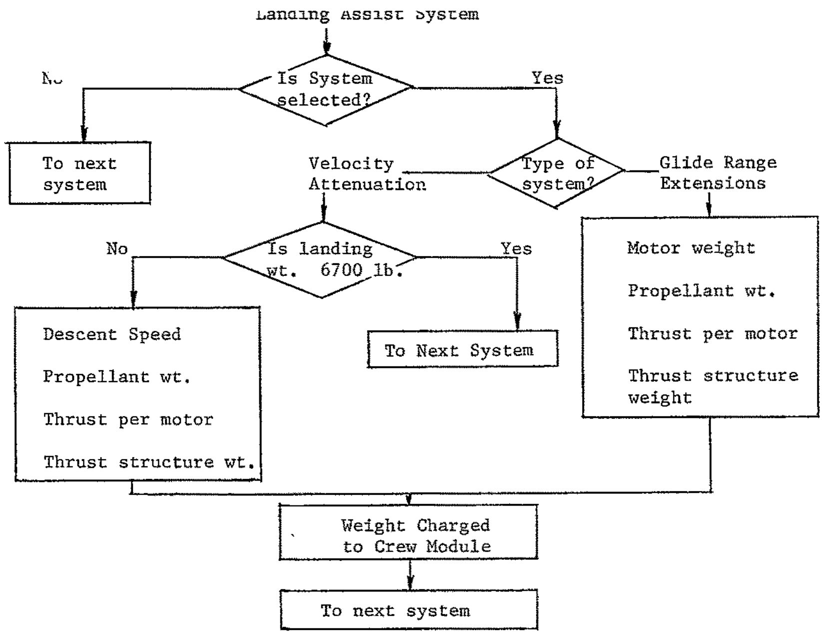


FIGURE 6.14-18



6.15 LAND SUBROUTINE - This subroutine computes the weight of various landing systems. Included are:

Aerial Recovery
Ballistic Landing
Landing Gears
  Gemini - Type
  Horizontal
Parachutes
  Main Chute
  Back-Up Chute
  Emergency Chute
Miscellaneous

Each system has a selection indicator so the system may or may not be included in the landing system weight. The weight of each system is calculated with reference to the landing, entry, or abort weight of the vehicle.

6.15.1 Equation Discussion - The following nine equations are used to compute all of the landing or recovery system weights.

1. WARS = C1 * (.056 *LANDW +56)
2. WBLSC = C2*(5.31*LANDW**.555)
3. WGG = C3*(.0444+.000336*VTDG**2/GGG)*LANDW
4. WHG = C4*(.133*LANDW**.88)
5. WMC = C6*C5**2*(ENTRYW/VDMC**2)+C7*C5**3*(ENTRYW/VDMC**2)**1.5+C8
6. WBC = C13*(C10*C9**2*(ENTRYW/VDBC**2)+C11*C9**3*(ENTRYW/VDBC**2)
   **1.5+C12)
7. WEC = C14*(C10*C9**2*(ABORTW/VDBC**2)+C11*C9**3*(ABORTW/VDBC**2)
   **1.5+C12)
8. WBAEC + WBC+WEC
9. WMLS = C15 * LANDW**C16

These equations are for air snatch recovery, three types of landing gear, five types of aerodynamic decelerators, and miscellaneous small subsystem weights associated with these subsystems.

6.15.1.1 Equation 1 defines the weight of the aerial recovery system. It is derived from Advanced Design study data.

Equation 1: WARS = C1*(6055*LANDW+56)

C1 = DLA(1) = selection indicator =0. or 1.

LANDW = MP(61,7) = vehicle landing weight (lbs.)

6.15.1.2 Equation 2 defines the weight of the ballistic landing system. The derivation of this equation is based on the MRS study, and is summarized in Figure 6.15-1.

Equation 2: WBLSC = C2*(5.31*LANDW**.555)

C2 = DLA(2) = selection indicator =0, or 1.

LANDW = MP(61,7) = vehicle landing weight (lbs.)

6.15.1.3 Equation 3 defines the weight of the Gemini-type landing gear, and is based on the MRS landing systems study.

Equation 3: WGG = C3*(.0444+.000336*VTDG**2/GGG)*LANDW

C3 = DLA(3) = selection indicator =0 or 1.

VTDG = DLA(4) = velocity at touchdown (ft./sec.)

GGG = DLA(5) = impact G's for Gemini gear

LANDW = MP(61,7) = vehicle landing weight (lbs.)

6.15.1.4 Equation 4 defines the weight of the horizontal landing gear. It is derived from the data in Figure 6.15-2. It includes the weight of the gear, struts, and shock absorbers. The equation is increased 33% to account for gear back-up structure and gear design complexities.

Equation 4: WHG = C4*(.133*LANDW**.88)

C4 = DLA(6) = selection indicator =0. or 1.

LANDW = MP(61,7) = vehicle landing weight (lbs.)

6.15.1.5 This set of equations, 5, 6, 7 and 8, determines the chute weights. They are derived from the general weight equation:

$$W_c = K_2D^2+K_3D^3+K_4$$

where D is the chute diameter and is determined by the relationship between chute diameter and descent velocity:

$$\text{ENTR.WT.} = (1/2\rho V^2)\left(\frac{\pi D^2}{4}\right)(CD)$$

$$= K_D V^2 D \qquad V = \text{descent vel.}$$

$$\text{or } D = K_1 \frac{(\text{ENTR.WR})^{\frac{1}{2}}}{V}$$

FIGURE 6.15-1

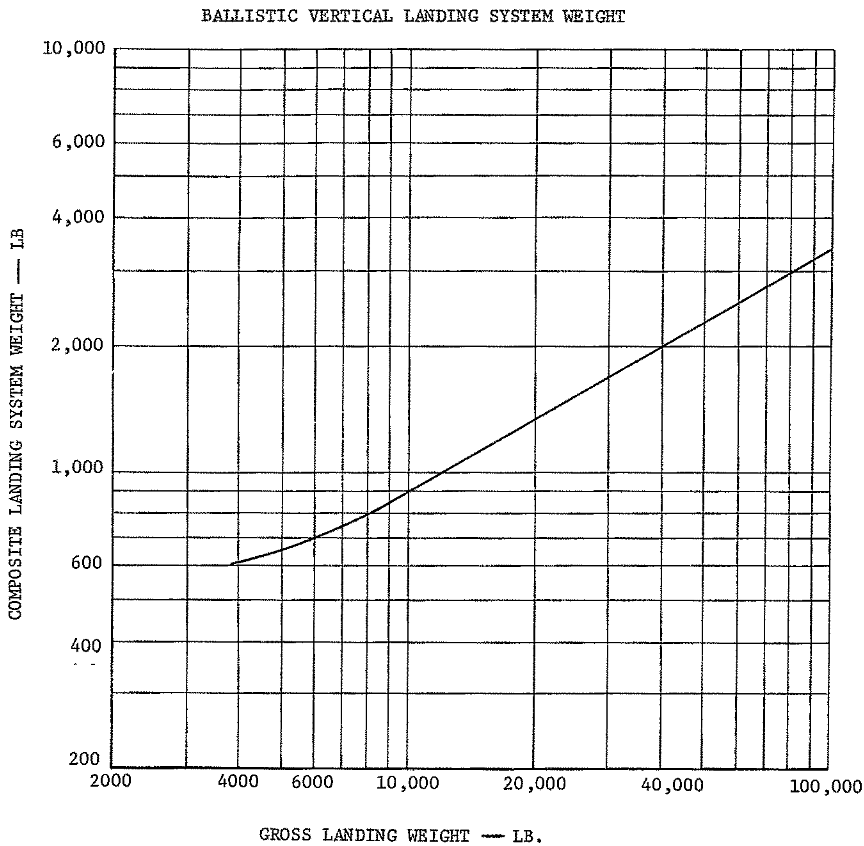
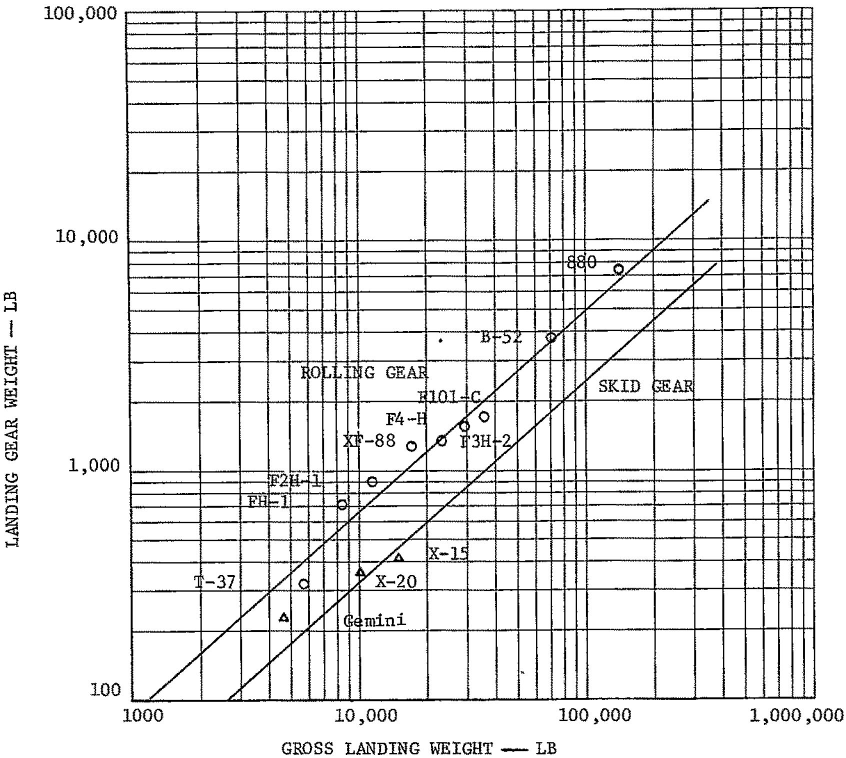

BALLISTIC VERTICAL LANDING SYSTEM WEIGHT
10,000
6,000
4,000
2,000
1,000
600
400
200
COMPOSITE LANDING SYSTEM WEIGHT — LB
2000
4000
6000
10,000
20,000
40,000
100,000
GROSS LANDING WEIGHT — LB.

FIGURE 6.15-2

HORIZONTAL LANDING GEAR WEIGHT



thus; $W_C = K_2K_1^2$ (ENTR.WT.)$V^{-2}+K_3K_1^3$ (ENTR.WT.)$^{3/2}V^{-3}+K_4$.
The constants K_1, K_2, K_3, and K_4 are determined by the type of chute used. Values for four chute types are listed below.

Chute Type		K_1	K_2	K_3	K_4	
	Item No. (Main)	C5	C6	C7	C8	
	Item No. (Back Up)	C7	C10	C11		C12
Ringsail		37.8	.02107	.000021	42.0	42.0
Glidesail		30.6	.02267	.000021	57.0	57.0
Parasail		30.1	.03567	.000037	57.0	57.0
Cloverleaf		18.0	.03445	.000036	60.0	60.0
Sailwing		28.5	.02267	.000008	215.0	90.0

Equation 5 defines the weight of the main chutes.

Equation 5: WMC = C6*C5**2*(ENTRYW/VDMC**2)+C7*C5**3*(ENTRYW/VDMC**2)**1.5+C8

C5 = DLA(7)
C6 = DLA(8)
C7 = DLA(9)
C8 = DLA(10)
Coefficients found in chart

ENTRYW = MP(61.6) = vehicle entry weight (lbs.)

VDMC = DLA(11) = descent vel. of main chute (ft./sec.)

6.15.1.6 Equation 6 defines the weight of the back-up chute.

Equation 6: WBC = C13*(C10*C9**2*(ENTRYW/VDBC**2) + C11*C9**2(ENTRYW/VDBC**2)**1.5+C12

C13 = DLA(17) = selection indicator =0. or 1.

C9 = DLA(12)
C10 = DLA(13)
C11 = DLA(14)
C12 = DLA(15)
Coefficients found in table

ENTRYW = MP(61,6) = vehicle entry weight (lbs.)

VDBC = DLA(16) = descent vel. of back-up chute (ft./sec.)

6.15.1.7 Equation 7 defines the weight of the emergency abort chute. It is the same as Equation 6, except the vehicle abort weight has been used in place of the vehicle entry weight.

Equation 7: WEC = C14*(C10*C9**2*(ABORTW/VDBC**2)+
C11*C9**3*(ABORTW/VDBC**2)**1.5+C12)

C14 = DLA(18) = selection indicator = 0. or 1.
ABORTW = MP(61,8) = vehicle abort weight (lbs.)

6.15.1.8 Equation 8 defines the weight of the back-up and emergency abort chutes.

Equation 8: WBAEC = WBC+WEC

WBC = Equation 6
WEC = Equation 7

6.15.1.9 Equation 9 defines the weight of the miscellaneous landing system. It is included to account for any weight not included in the first eight equations.

Equation 9: WMLS = C15*LANDW**C16

C15 = DLA(19) = coefficient for misc. land. sys.
LANDW = MP(61,7) = vehicle landing weight (lbs.)
C16 = DLA(20) = exponent for misc. land. sys.

6.15.2 Logic Flow Diagram - The logic flow of this subroutine is shown in Figure 6.15-3.

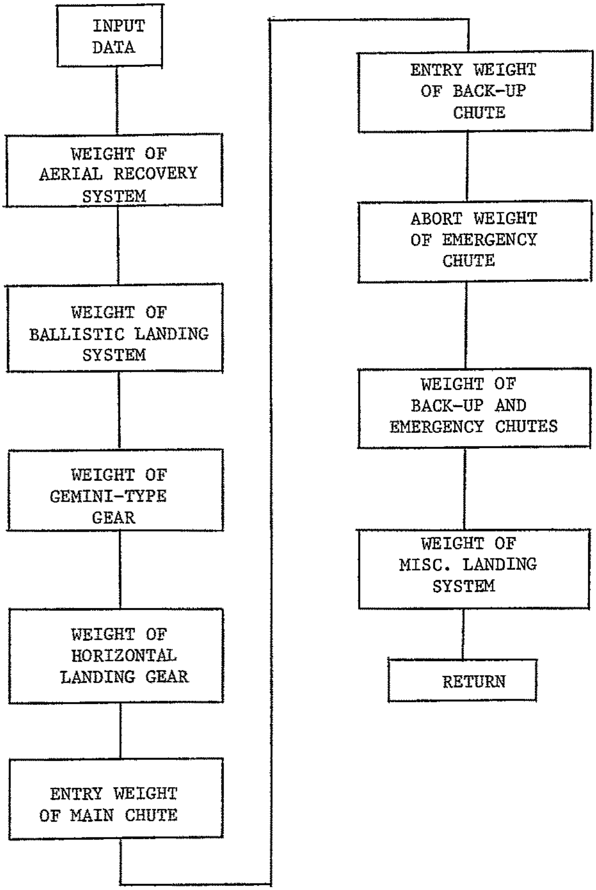
6.16 SIZE SUBROUTINE - The purpose of this routine is to find the minimum overall length (OL) for the vehicle which satisfies all of ten possible sizing requirements. Any variation from the size of the original vehicle is done by ballooning. That is, the entire vehicle is photographically enlarged or reduced until it meets the particular sizing requirement.

For each of the ten possible sizing requirements to be met, an overall length for the vehicle which meets this requirement is calculated and saved. After all ten of the possible overall lengths have been calculated, the largest of these is selected. The overall length of the entry vehicle is assigned this volume which is stored for all supporting subroutines to use.

FIGURE 6.15-3

LAND SUBROUTINE

FLOW DIAGRAM



The first sizing requirement to be met is entry vehicle volume. The inner volume of the vehicle is calculated based on the original read-in overall length. If this volume does not satisfy the current volume requirement, an iteration procedure is begun to find the necessary overall length. Two to four iterations are generally sufficient for convergence. The resulting overall length is saved.

Next the maximum floor to ceiling height is found based on the original length of the vehicle. If this maximum height does not meet the required maximum height, then the vehicle is sized accordingly and the resulting overall length is saved.

There are eight other floor to ceiling height tests that can be made. That is, at each of eight different body stations, a floor to ceiling height may be specified. For each case, a new vehicle overall length is determined to meet the particular height requirement. Each of these eight overall lengths is saved. The final value for the overall length of the vehicle is taken to be the largest value of the ten overall lengths calculated.

Finally, the mission module is sized by the same method that is used in the GEOM subroutine.

The logic flow is shown in Figure 6.16-1.

6.17 GENERATE - The Generate subroutine was developed to provide the Cost Module with certain inputs that are not available in the other Sizing Module subroutines. The calculation flow diagram for Generate is presented in Figure 6.17-1. The Cost Module structural CER's are sensitive to type of construction, differences between thermal protection, aerodynamic control surfaces, and basic structure, and the differences between the crew section and the cargo/propulsion section. Generate is used to provide this information to the Cost Model.

Generate performs three basic functions for the thermal protection, aerodynamic control surfaces, and the basic structure subsystems.

(1) Determines the type of material by testing the Sizing Module input material selectors. The material for thermal protection is determined by checking the body temperature of the vehicle at different stations.

(2) Determines weights and areas by summing various Sizing Module outputs.

(3) Segregates the materials and areas between the crew section and the cargo/propulsion section. Since the cargo propulsion section begins at the forward location of the cargo, this point is used to determine the split between the crew section and the cargo/propulsion section. The Generate logic determines where the cargo/propulsion section begins and then allocates the weights to the crew section or the cargo/propulsion section. The ratio of section length to total length is used to prorate those weights that are in both the crew section and cargo/propulsion section but that are calculated in total in the other Sizing Module subroutines.

FIGURE 6.16-1

SIZING ROUTINE

LOGIC FLOW

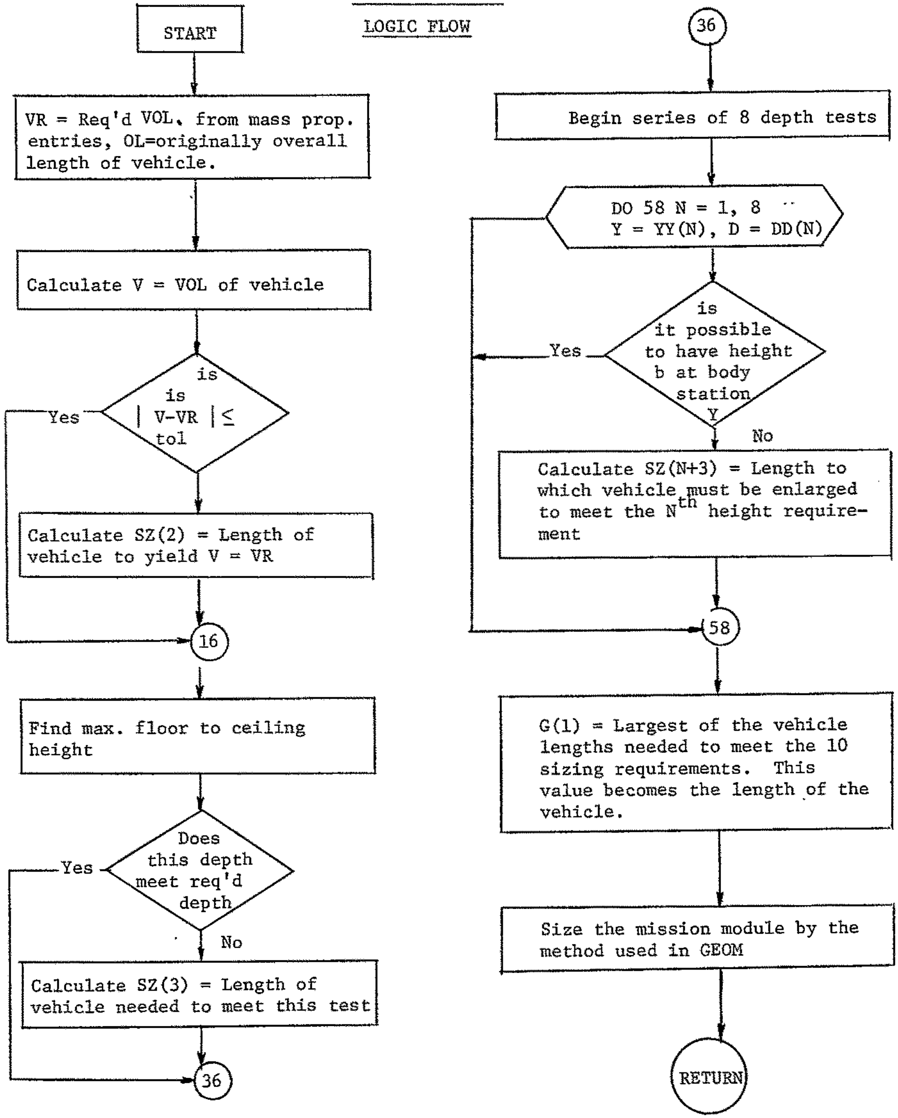


FIGURE 6.17-1

GENERATE SUBROUTINE - FLOW LOGIC

INPUTS SIZING MODEL | FROM OTHER SUBROUTINES

Test forward location of cargo to determine crew section - cargo/propulsion section division.

Total the wetted areas of the aerodynamic control surfaces.

Split the mounting structure between the two sections.

Test material selectors and allocate the weight by section for bulkheads, windows, hatches, floors, and finish.

Allocate shell weight and ring weight based on location and material selectors.

Test the temperature at each body station to determine the type and weight of material for thermal protection.

Determine the type and weight of material for the aero. Control surfaces by testing the appropriate material selectors and the vehicle location of the surfaces.

Return

SPACECRAFT COST MODEL
AND INTERFACE

7.0 SPACECRAFT COST MODEL AND INTERFACE - The Spacecraft Cost Model contains all of the Cost Estimating Relationships (CER's) necessary to estimate the total spacecraft cost. The Cost Model also adds the launch vehicle cost to the spacecraft cost to obtain the total program cost. The Interface subroutine is necessary to collect and prepare outputs from the Sizing Model, Inventory Model, and Executive Logic for use in the Spacecraft Cost Model.

7.1 SPACECRAFT COST MODEL - The CER's that are utilized in the Spacecraft Cost Model are presented and discussed in Volume II, Books 3 and 5 of this report. The Cost Model simply calculates the results of each CER and provides the desired subtotals and cost summaries that the user selects. Any one or all of the following cost summaries are available to the user. See Section 4.3 for sample output summaries.

1. Summary 1 - One page gross cost summary providing spacecraft and launch vehicle cost by program phase.
2. Summary 2 - Summary 1 expanded to three pages to provide cost by subsystem groups and project segment.
3. Summary 3 - Summary 1 expanded to three pages to provide cost by cost category (e.g., engineering, tooling, etc.) and program phase.
4. Summary 4 - This is a complete detailed output of each CER that is programmed for the spacecraft.
5. Summary 5 - Alphabetical listing of the symbols and input values associated with the estimated spacecraft cost.

The calculation sequence of the Spacecraft Cost Model attempts to follow the Cost Element Structure (CES) and the order in which the CER's have been presented in Volume II, Books 3 and 5. Figure 7-1 presents the basic Spacecraft Cost Model calculation logic.

The first unit hardware costs are calculated first since their results are used extensively throughout the RDT&E, Investment, and Operational phases.

The RDT&E Phase is calculated next and is grouped by project segment. The entry vehicle and mission module design and development costs are calculated first followed by the support and integration segments. The system integration segment is a relatively large area and therefore is separated by type of test hardware and test operation.

SPACECRAFT COST MODEL CALCULATION LOGIC

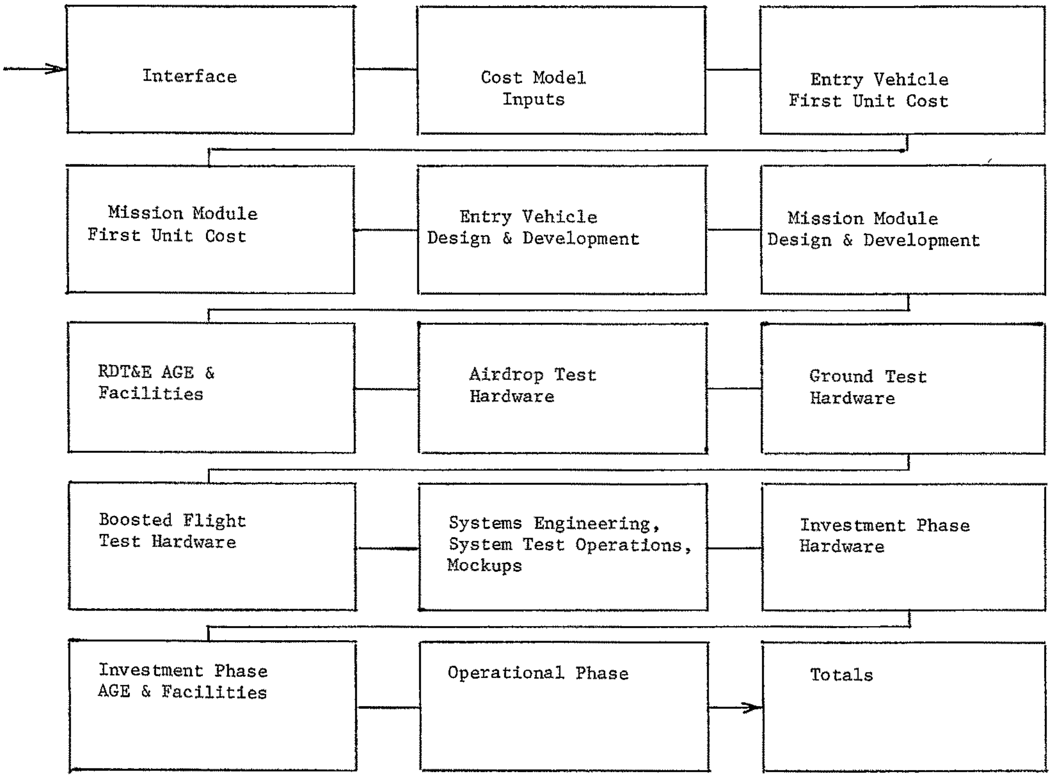


FIGURE 7-1

The Investment Phase is calculated next. The CER's here are simply the application of the appropriate learning curves and hardware quantities to first unit cost.

The Operational Phase is calculated next.

There are a few exceptions to this general calculation sequence such as first unit sustaining engineering cost which is a function of design and development cost and therefore must be calculated after the design and development cost. These calculations are minor and are performed between the major subroutines but not shown on the logic diagram.

The last subroutine is the summation of the detail costs that are required for the various output summaries.

7.2 INTERFACE - The interface is a prepatory subroutine for the Spacecraft Cost Model inputs. Since the Spacecraft Cost Model has the capability of estimating many different vehicle configurations, the Interface is required for testing and assembling the outputs of the Sizing Model, Inventory Model, and Executive Logic to assure the proper inputs to the Spacecraft Cost Model. The inputs to the Interface are tested and assembled to be consistent with the study ground rules, user inputs, and established vehicle configurations. For this reason, the Interface is sensitive to, and constrained by the study ground rules, vehicle configurations, and general program definitions.

The Sizing Model presents the weights in the normal format for weight summaries and not the grouping of weights as used by the Cost Model. The Interface collects these values and assembles them as utilized in the Cost Model. Some inputs required by the Cost Model are internal calculations in the Sizing Model and cannot be passed to the Interface. These internal calculations are simply repeated in the Interface to obtain the desired values.

A series of switches and tests are provided to establish the Cost Model inputs consistent with the study ground rules, user inputs, and vehicle configuration.

The nine calculation sections that make up the Interface are shown in Figure 7-2 and are defined below.

7.2.1 INITIAL - The purpose of this section is to establish all testing switches and internal values that are used in the following sections of the Interface. There are seven different sets of calculations in the Initial Section. Because these calculations are internal to Interface and are used in more than one section, they are grouped together in the Initial Section. These calculations are summarized below.

1. The weight of the Telecommunications subsystem is input into the Sizing Model based on the technical concept desired. Since the cost is a function of the type of concept, the Sizing Model output weight and the technical definition corresponding with that weight are used to determine the value of the telecommunications concept switch.

2. The structural CER's for an integral configuration require that the structural weight and material distribution be separated into the crew section and the cargo/propulsion section. This is basically accomplished by the Sizing Model in a special subroutine called Generate. This portion of the Inital further assembles these structural values as required by the Cost Model. The modular vehicles consists only of a crew section. However, because of the method used to segregate the sections in Generate, a small amount of length and weight will be allocated to the cargo-propulsion section for modular vehicles. For this reason, if there is a modular vehicle, the cargo/propulsion section weights and areas are added to the crew section parameters.

3. Certain costs, weights, and subsystem concepts which are used in the following sections are dependent on configuration type and reuse mode. The user inputs are used to set the switch which reflects which configuration and reuse mode are being used.

4. The Sizing Model segregates the docking subsystem structural weight. Since this is a part of structure and costed as such, the weight is added to the basic structural weight in Generate. If the vehicle is an integral lifting body, the internal integral propulsion tank is a part of the vehicle basic structure, and its weight is added to the basic structural weight in Generate.

INTERFACE CALCULATION LOGIC

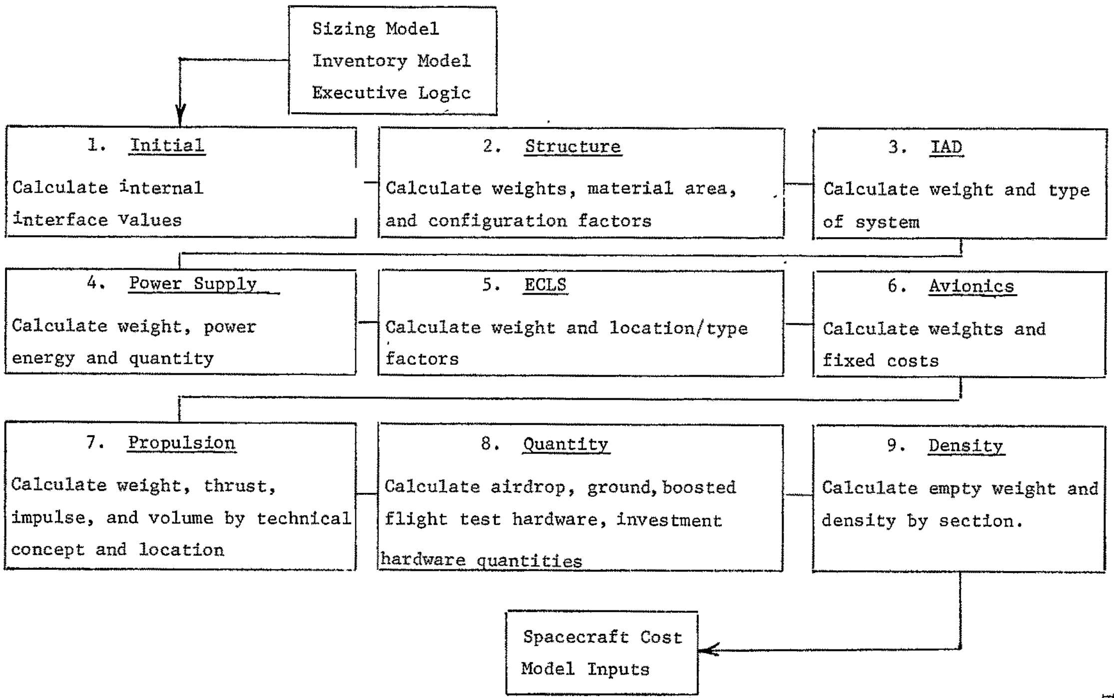


FIGURE 7-2

5. The volumes, access areas, and total wetted areas by section are internal values that are used in Structure and Miscellaneous to calculate the area factors and density factors. The areas and volumes are Sizing Model outputs.

6. The type of innerbody construction is used in calculating certain material complexity factors. A structural Sizing Model input is used to determine the value of the construction switch.

7. The propulsion engine calculations are dependent on the engine cooling method, and the tank equations are dependent on the quantity of tanks per system. These inputs are not used in the Sizing Model, so they are defined and used in the Interface. The temporary values and switches are set in Initial for use in the Propulsion section.

7.2.2 STRUCTURE - The Thermal/Structure subsystem group costs are functions of weight, configuration, density, area, and material factors. The structure is divided into basic structure, thermal protection, and aerodynamic control surfaces. Each of these divisions is further segregated between the crew section and the E/V cargo/propulsion section. Depending on the type of cost, First Unit or Design and Development, and the labor category, the CER's estimate the structural costs at various levels of structural division. The individual weights, which are Generate outputs, are grouped according to the level of structural division needed for a CER. The area factors are calculated based on the areas from Initial and their definitions as presented in Volume II, Books 3 and 5. The material factors are functions of weight and the material complexity factors from Table 6-1 in Volume II, Books 3 and 5.

7.2.3 INFLATABLE AERODYNAMIC DEVICES - These CER's are a function of subsystem weight, spacecraft weight at recovery subsystem deployment, and type of subsystem. A parachute subsystem is used for all vehicles except a IA (land) or a IB vehicle which uses a sailwing subsystem. The appropriate weights are directly assigned to the proper subsystem after the type of vehicle is determined.

7.2.4 POWER SUPPLY - The Power Supply/Ordnance subsystem group CER's are dependent on weight, power, energy, and quantity. The battery and ordnance weights and the power and energy are direct Sizing Model outputs. The weights of the fuel cell, electrical distribution, reactant supply, and hydraulic and pneumatic subsystems are calculated using the Sizing Model equations since these are internal calculations in the Sizing Model. The quantity of batteries is calculated based on the battery weight and the specific weight per battery.

7.2.5 ECLS - The number of men, mission duration, and weights are used in the ECLS CER's. These inputs are all obtained directly from the Sizing Model outputs. Since the ECS subsystem CER's are sensitive to the differences between a cryogenic and a storable subsystem, a factor is defined to reflect these differences.. The weights of the different types of components are used to determine if there is a cryogenic or storable subsystem. The location of the equipment, either in the E/V or the M/M is determined based on Sizing Model output weight, and the percentage split between E/V and M/M is incorporated in the above factor.

7.2.6 AVIONICS - This subsystem group uses the weights and the First Unit and Design and Development costs previously defined in Section 6, Volume II, Books 3 and 5. The weights are direct outputs of the Sizing Model. The Telecommunications and Guidance and Control subsystems First Unit and Design and Development Material, CFE, and Subcontract costs are dependent on the technical concept, the type of vehicle, and the reuse mode being considered. The type of concept is defined in the Initial section. The vehicle configuration and reuse mode determine which costs are used from Table 6-11 in Section 5.1.4.9 of Volume II, Books 3 and 5.

7.2.7 PROPULSION - The numerous Propulsion CER's are functions of thrust, volume, quantity, total impulse, and weight. Although there are many calculations in Propulsion, there are few basic equations. The Cost Model inputs are based directly on the Sizing Model outputs, but the location and use of components are based on predetermined propulsion subsystem combinations. The Entry Attitude Control, Vernier Maneuver, and Main Orbital Maneuver subsystem inputs are determined by the same general methodology. Certain

Sizing Model inputs are subsystem locators and switches. These inputs are used together with the propulsion switches defined in Initial to determine where and when subsystem components are used. The general methodology is as follows:

1. Determine the location of the subsystem.
2. Determine type of engines being considered using the cooling method switches from Initial.
3. Determine if additional engines, tanks, or lines and valves are required for the subsystem to perform additional propulsion functions.
4. Assign the correct Sizing Model outputs to the Cost Model inputs.

The Solid Deorbit Rocket Motor, Landing Assist, and Launch Escape Motor subsystems use Sizing Model locators to determine where the subsystem is being used. The total impulse per motor is calculated using the internal Sizing Model equations. The Launch Upper Stage Boost subsystem uses the cooling method selector and Sizing Model outputs. Since certain CER's are sensitive to the type and weight of propellant, these inputs are determined through Sizing Model input selectors and output weights.

7.2.8 QUANTITY - Various quantities are used in the Cost Model to calculate recurring hardware costs. According to study groundrules, the boosted flight test program consists of five vehicles and five flights. The quantities of investment hardware are passed directly from the Inventory Model. The equivalent sets of AGE are calculated based on operational indicators from the Executive Logic. The numbers of equivalent sets of ground test hardware are inputed directly based on their definition and quantities. Since certain definitions are dependent on configurations and/or reuse mode, the correct value is determined after the type of vehicle and reuse mode is determined.

7.2.9 Miscellaneous - Certain structural CER's are dependent on vehicle density. The appropriate Sizing Model output weights are grouped and used with the volumes from Initial to calculate the density factors. Because of the methods of calculation in a few areas, the Sizing Model will calculate a very small weight (order of magnitude of 10^{-2}) for a nonexistent subsystem. Since these subsystems should not be costed, all thrusts and weights are tested, and if their value is less than one, it is set equal to zero.

8.0 LAUNCH VEHICLE COST MODEL

8.1 LAUNCH VEHICLE USER INPUTS - The launch vehicle cost model for OCPDM uses five different launch vehicles. The program user inputs the launch vehicle type with LVT.

LVT = 1 - expendable first and second stages
solid/liquid
(LO_2/LH_2)

LVT = 2 - expendable first and second stages
liquid/liquid
(LO_2/RP - LO_2/LH_2)

LVT = 3 - expendable first stage
solid (260" dia.)

LVT = 4 - expendable first stage
liquid (LO_2/RP)

LVT = 5 - reusable first stage
VTOHL (LO_2/LH_2)

Launch vehicles LVT = 1 and LVT = 2 are only used with vehicles without integral upper stage propulsion (REUSE = 1, 2 and 3). Launch vehicles LVT = 3 and LVT = 4 are used with the integral upper stage propulsion vehicles, i.e. REUSE = 4 and 5. The reusable first stage is only used with the REUSE = 6 entry vehicle. Table 8-1 summarizes the compatible launch vehicles and reuse categories.

TABLE 8 - 1

LAUNCH VEHICLE TYPE	CAN ONLY BE USED WITH
LVT = 1 LVT = 2	REUSE = 1, 2 or 3
LVT = 3 LVT = 4	REUSE = 4 or 5
LVT = 5	REUSE = 6

If the program user has selected his reuse category he may want to find the least expensive launch vehicle applicable to that category. Table 8-1 illustrates which launch vehicles are applicable for any reuse category. If the program user inputs LVT = 0 the program will cost all the launch vehicles applicable to that reuse category and return the cheapest one. A launch vehicle choice is available for every reuse category except REUSE = 6.

Other user inputs directly related to the launch vehicle are:

LVTWE - launch vehicle throw weight in pounds for a due east launch from ETR.

PØMS(1)- probability of mission success

STGVEL - staging velocity in feet/second for the first stage (only used for LVT = 3)

8.2 COST MODEL - The launch vehicle cost model breaks the launch vehicle costs into four phases:

a) contract definition
b) RDT & E
c) investment
d) operations

Within each phase the costs are broken down into:

a) basic launch vehicle cost
b) fee
c) office management

The CER's for the basic costs, fee and office management costs for RDT & E, investment and operations are shown in Volume II, book 5, section 7. The contract definition costs are defined using the basic launch vehicle development cost as follows:

a) basic contract definition cost is 1% of the launch vehicle development cost
b) contract definition fee is 10% of the basic contract definition cost
c) contract definition office management is 10% of the basic contract definition cost, i.e. fee = office management.

Office management cost for the investment and operations phases is calculated in total as a function of the average operations cost. This cost is broken down with 45% for the investment phase and 55% for the operations phase. The program limits the average office management cost to 87% of the average operations cost.

The detailed explanation of the CER's in the launch vehicle cost model is given in Volume II, book 5, section 7.

8.3 SIGNIFICANT VARIABLE NAMES FOR THE LAUNCH VEHICLE COST MODEL

ABAR	true annual launch rate
ALVIN	average launch vehicle investment cost
ALVØP	average launch vehicle operations cost
LVC (I, 1)	launch vehicle development cost
LVC (I, 2)	launch vehicle office management cost for the development phase
LVC (I, 3)	fee for the launch vehicle development
LVC (I, 4)	total launch vehicle investment cost
LVC (I, 5)	launch vehicle office management cost for the investment phase
LVC (I, 6)	fee for the launch vehicle investment
LVC (I, 7)	launch vehicle operations cost
LVC (I, 8)	launch vehicle office management cost f the operations phase
LVC (I, 9)	fee for the launch vehicle operations
LVC (I, 10)	total launch vehicle costs
LVT	launch vehicle type indicator (user input)
LVTWE	launch vehicle throw weight-due east ETR launch (user input)
NØAL	total number of attempted launches
P	launch vehicle degradation factor for the launch vehicle throw weight capability
PL	program length in years (user input)
SCWT	total spacecraft weight in pounds
STGVEL	staging velocity of the first stage in thousands of feet per second (user input)

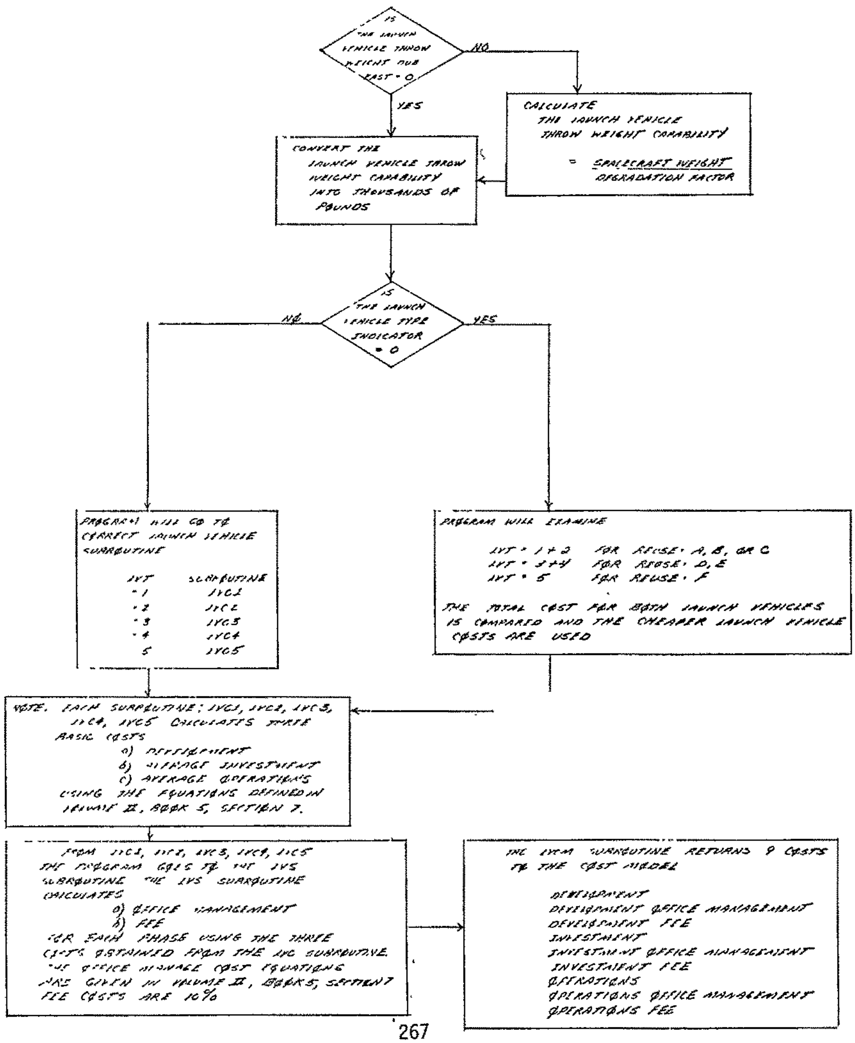
TØM	launch vehicle office management costs for the investment and operations phases
TW	spacecraft weight converted to a due east ETR launch (pounds)
WGT	spacecraft TW weight in thousands of pounds

8.4 Launch Vehicle Logic Flow - The launch vehicle cost model logic flow is illustrated in Figure 8-1.

LAUNCH VEHICLE COST MODEL

LOGIC FLOW

FIGURE 8-1



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9. INVENTORY MODEL - The inventory model for OCPDM study provides the cost model with the following information:

a) Number of successful missions required to deliver the cargo and men at the required rate.
b) Number of launches required to meet the successful mission rate.
c) Number of entry vehicles required to sustain the launch rate.
d) Number of cargo/propulsion sections required to sustain the launch rate.
e) Number of launch vehicles required to sustain the launch rate.

9.1 LAUNCH REQUIREMENTS - In determining the inventory quantities required to support an operational program both the number of successful missions (NOSM) and the number of attempted launches (NOAL) are obtained. Each of these values is a function of two user inputs to the OCPDM model. The number of successful missions is defined by:

9.1-1 $$NOSM = TCW/CWL$$

where

TCW - Total cargo weight in pounds delivered to orbit for an entire program

CWL - Cargo weight/launch in pounds

The number of attempted launches is defined by:

9.1-2 $$NOAL = LR/PL$$

where

LR - Annual launch rate

PL - Operational program length in years

In the actual use of the program all the variables, i.e. TCW, CWL, LR and PL are usually not defined for the same run. Either the number of successful missions is known and the number of attempted launches unknown or the reverse situation holds. The inventory model uses the mission reliability to define the relationship between NOSM and NOAL and consequently if NOSM is known, NOAL can be found or if NOAL is known, NOSM can be found.

9.1.1 NUMBER OF ATTEMPTED LAUNCHES - This subsection will define the technique used to compute NOAL when NOSM is known. The basic mathematical reasoning for the inverse case, i.e. NOAL known and NOSM unknown is identical and will be covered in section 9.1.2.

The total number of launches, NOAL, is the sum of the successful missions, NOSM, plus those launches which do not result in a successful mission, i.e. delivery of cargo and/or men. Stated in another way, how many launches should be expected when it is necessary to have NOSM successful missions. The program user is required to input the mission reliability, for both the spacecraft and the launch vehicle. The total mission reliability, MR, is the product of these two reliabilities.

Clearly if MR = 1.0, then every launch results in a successful mission and the number of launches would equal the number of successful missions.

In the real case the mission reliability is less than unity and consequently there will be some launches which will not result in a successful mission. Accordingly the total number of launches will exceed the number of successful missions.

In order to calculate the expected number of total launches the following assumptions are used:

a) Each launch is an independent event, i.e. the success or failure of the previous launches does not affect the mission success of the next launch.
b) The mission reliability remains constant throughout the entire launch program.
c) Only two outcomes are considered for each launch, i.e. either mission success or mission failure.

The above assumptions satisfy the conditions for Bernoulli trials and the probability of obtaining NOSM successful missions in exactly NOAL total launches can be expressed by the negative binomial distribution, i.e.

$$9.1\text{-}3 \quad P(NOAL;\ NOSM) = \binom{NOAL-1}{NOAL-NOSM} (MR)^{NOSM} (1-MR)^{NOAL-NOSM}$$

$$0 \leq MR \leq 1$$

$$NOAL \geq NOSM$$

The probability of having NOSM successful missions in exactly NOSM launches becomes

$$P(NOSM; NOSM) = \binom{NOSM-1}{o} (MR)^{NOSM} (1-NOSM)^{o}$$

$$= (MR)^{NOSM}$$

This is usually an extremely small number except for small values of NOSM or very high values of MR. The difference, NOAL-NOSM, represents the number of failures which will occur before the NOSM th successful mission. Equation 9.1-3 can be rewritten in terms of the number of successful missions, NOSM, and the number of failures, f. f = NOAL-NOSM

$$9.1\text{-}4 \quad P(f;NOSM) = \binom{NOSM + f - 1}{f} (MR)^{NOSM} (1-MR)^{f}$$

Both equation 9.1-3 and 9.1-4 indicate the probability of achieving NOSM successful missions in exactly NOAL or NOSM + f launches.

The probability of achieving NOSM successful flights in NOAL or fewer launches becomes

$$\text{Alpha} \leq P(NOAL;NOSM) + P(NOAL\text{-}1;NOSM) + \ldots + P(NOSM;NOSM)$$

$$9.1\text{-}5 \quad \text{Alpha} \leq \sum_{i = NOSM}^{NOAL} \binom{i-1}{i-NOSM} (MR)^{NOSM} (1-MR)^{i-NOSM}$$

Equation 9.1-5 is used within the OCPDM inventory model to find the number of launches, NOAL, for which the probability of obtaining NOSM successful launches is at least ALPHA. The probability or confidence level ALPHA equals .90 (90%) in the OCPDM program. This value along with the number of successful missions and the mission reliability are used to find the corresponding number of total launches.

In figure 9-1 the use of equation 9.1-5 is illustrated for NOSM = 100 missions and several MR values.

If MR =.96 and a 90% ALPHA value is selected then 107 launches should be planned for. In other words, there is a probability of .90 that 100 successful missions will be obtained in the first 107 launches.

9.1.2 NUMBER OF SUCCESSFUL MISSIONS - All of the assumptions described in section 9.1.1 are still valid for this derivation. The only change is that NOSM is now the unknown variable and NOAL the known. Equation 9.1-3 is still valid

$$P(NOAL;NOSM) = \binom{NOAL-1}{NOAL-NOSM} (MR)^{NOSM} (1-MR)^{NOAL-NOSM}$$

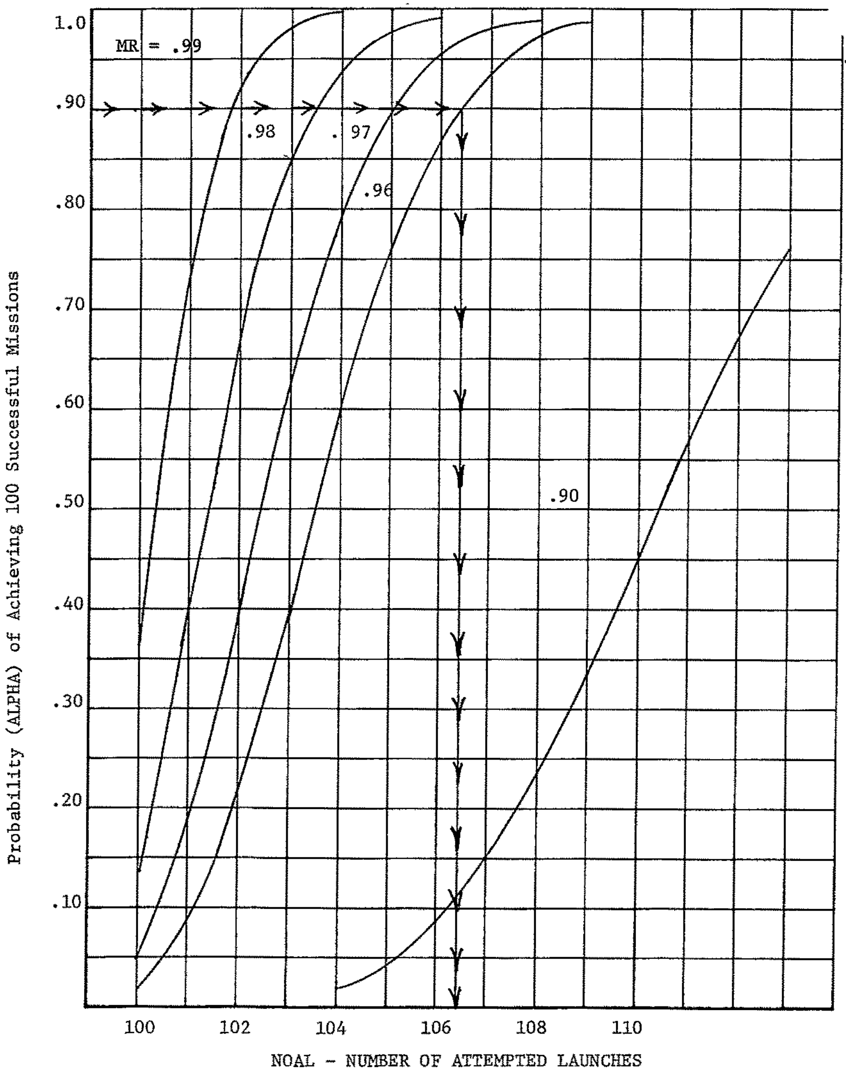
$$0 \leq MR \leq 1$$

$$NOAL \geq NOSM$$

and is still defined as the probability of obtaining exactly NOSM successful missions in NOAL launches. In section 9.1.1 we held NOSM constant and iterated NOAL up until the sum of the NOAL-NOSM + 1 terms exceeded the .90 level (90% confidence).

Figure 9-1

PROBABILITY OF ACHIEVING 100 SUCCESSFUL FLIGHTS



To find the number of successful missions for a fixed number of attempted launches, NOAL is held constant and NOSM is iterated down until the sum of the terms exceeds .9 (90% confidence). The terms are:

$$\text{ALPHA} \leq P(\text{NOAL};\text{NOAL}) + P(\text{NOAL};\text{NOAL}-1) + \ldots + P(\text{NOAL};\text{NOSM})$$

or

$$\text{9.1-6} \quad \text{ALPHA} \leq \sum_{i\,=\,\text{NOAL}}^{\text{NOSM}} \binom{\text{NOAL}-1}{\text{NOAL}-i} (\text{MR})^{i} \, (1-\text{MR})^{\text{NOAL}-i}$$

Equation 9.1-6 is analagous to equation 9.1-5 in section 9.1.1.

9.2 <u>ENTRY VEHICLE INVENTORY</u> - The inventory model which is described below uses a dynamic programming approach. The entry vehicle inventory is divided into three categories:

a) Abort vehicles - the number of vehicles which will be lost during the program because their launch to launch reliability, POSR, is less than unity. The number of these vehicles is designated by NA.

b) Pipeline vehicles - entry vehicles which added to the inventory in order to sustain the launch rate, i.e. keep the pipeline completely filled at all times. Quantity is designated by NL.

c) Additional vehicles - number of entry vehicles which are added to the inventory because of the design life limitations. Quantity is designated by NR. The pipeline and additional vehicles are collectively designated as "no-loss" vehicles, i.e. since the expected number of abort vehicles is calculated, it is assumed that all other vehicles will be capable of their full design life.

The dynamic programming approach is used because of the obvious interactions between the quantities of vehicles described in a, b and c above. This approach does not calculate a quantity for a, b and c and merely add them together. Rather as each quantity is calculated, the current total is used in determining the inventory quantity for the next category. Specifically in the OCPDM inventory model the following logic is used.

a) Calculate number of launches - NOAL

b) Calculate number of abort vehicles - NA

c) Calculate number of pipeline vehicles - NL

d) Calculate number of additional vehicles - NR

As the model proceeds from b to c to d, a current inventory total, NOV (2) is calculated which is the sum of all the vehicles which have been added up to that point in the logic. Obviously after b above, the current inventory total equals the number of abort vehicles, but by the time the model has gone through 'd' the inventory total includes abort, pipeline and additional vehicles. Several references are made to the current inventory total in the following sections and, in all cases, it is the sum of all vehicles added to the inventory up to that time in the logic.

9.2.1 NONREUSABLE VEHICLES - The OCPDM inventory model considers both reusable and nonreusable entry vehicles. The total inventory for nonreusable vehicles is set equal to the number of launches. This total is broken down into the number of abort vehicles and the number of additional vehicles. Since all vehicles are expendable, there is no pipeline condition and consequently no pipeline vehicles. For the nonreusable case, the definition of an abort is more restricted than the general definition given in section 9.2. For nonreusable vehicles, a vehicle loss can only occur during the mission phase, i.e. launch to recovery, as compared with the general definition of launch to launch reliability. After the number of abort vehicles has been calculated, the number of additional vehicles is the total number of launches minus the number of abort vehicles.

9.2.2 REUSABLE VEHICLES

9.2.2.1 ABORT VEHICLES - As indicated in section 9.2 the first inventory quantity calculated is the number of abort vehicles. For reusable vehicles, the general definition is used, i.e. an entry vehicle is considered aborted if during either the mission, transportation, recertification or prelaunch phase the vehicle is rendered unfit for any future launch. This launch to launch reliability is a user input to the program.

The calculation of the number of abort vehicles, NA, depends on only three parameters.

a) POSR - launch to launch reliability

b) NOAL - number of launches

c) ALPHA2 - confidence level (%) on NA

The following assumptions are made:

a) Each launch to launch period is an independent event

b) The launch to launch reliability remains constant

These assumptions allow the use of the binomial theorem, i.e. the probability of having exactly NA aborts in NOAL launches is given by:

$$9.2\text{-}1 \quad P(NA;NOAL, POSR) = \binom{NOAL}{NA} (POSR)^{NOAL-NA}(1-POSR)^{NA}$$

$$0 \leq POSR \leq 1$$

$$NA \leq NOAL$$

For example, assume NOAL = 100 launch and POSR = .99, then the probability of having no aborts becomes

$$P(0;100, .99) = \binom{100}{o} (.99)^{100} (1-.99)^{o}$$

$$= (.99)^{100} \cong .37$$

The probability of no losses is only 37%. The probability of one or more losses is 63%.

The probability of have NA or fewer aborts equals

$$9.2\text{-}2 \quad ALPHA2 = P(NA;NOAL, POSR) + P(NA-1; NOAL, POSR) + \ldots + P(0;NOAL, POSR)$$

In the OCPDM model ALPHA2 is set at 90%, i.e. there is only a 10% probability that the calculated number of aborts will be exceeded.

9.2.2.2 PIPELINE VEHICLES - Pipeline vehicles refer to the vehicles which must be in the launch to launch cycle to support the launch rate. The launch to launch cycle consists of four phases:

a) Mission

b) Transportation

c) Recertification

d) Prelaunch

The mission is further divided into three phases:

a) Ascent (1 day)

b) Orbit stay time

c) Return time (1 day)

The orbit stay time is a user input. The transportation time is a function of the transportation mode and the reusability of the entry vehicle, both of which are also user inputs. First unit recertification time and first unit prelaunch time are calculated in the flow time subroutine REFTIM which is described in section 14. A unit learning rate of 90% is applied to the recertification time and a rate of 87.8% is used for prelaunch time.

The launch to launch cycle determines the number of pipeline vehicles required at any time. The turnaround time (launch to launch) at the nth launch is

9.2-3 $TTIME = MT + TT + TCD\ (n-NOV(2))^{(-.152)} + TPL\ (n)^{(-.188)}$

TTIME = turnaround time
MT = mission time
TT = transportation time
TCD = recertification time (first unit)
TPL = prelaunch time (first unit)
NOV(2) = current inventory

The learning in recertification does not begin until after the first NOV(2) flights. The number of pipeline vehicles, NPLV for any TTIME is given by:

9.2-4 $NPLV = (TTIME) \times (ABAR)/365$

ABAR = true annual launch rate

The calculation of the number of pipeline vehicles in the OCPDM inventory model occurs in three steps:

a) Calculate minimum turnaround time and the number of pipeline vehicles required.
b) Calculate maximum turnaround time and the number of pipeline vehicles required.
c) Calculate the launch numbers at which the number of pipeline vehicles decreased by one. Calculate the probability that the aborted vehicles will deplete the inventory below pipeline requirements.

The minimum turnaround time is calculated using NOAL launches and the current inventory total which still equals NA. This minimum turnaround time is used in 9.2-4 to calculate the pipeline requirement. Since it is assumed that all NA vehicles will be lost by the NOAL th flight, all the minimum pipeline vehicles, NL, are added to the current inventory.

The maximum turnaround time is calculated for the NOV(2)th launch to launch period since it is assumed that each of the first NOV(2) launches will use a new vehicle and no pipeline problem can exist. The number of pipeline vehicles is calculated for the maximum turnaround time. If the current inventory equals or exceeds the number of maximum pipeline vehicles, no vehicles are added in this step. If the current inventory is less than the number of maximum pipeline vehicles then one vehicle is added to NL, the current inventory increases by one and a new maximum turnaround time is calculated. The current inventory is iterated and compared with the maximum pipeline requirement until they are equal.

The third step in obtaining the total pipeline inventory examines the pipeline requirements between the maximum and minimum turnaround times. The turnaround time for each launch decreases because of the learning effect on recertification and prelaunch times. As the turnaround time decreases the number of pipeline vehicles also decreases, although while the turnaround time is a continuously decreasing quantity, the pipeline quantity decreases in steps as illustrated in figures 9-2 and 9-3. This final step in obtaining the total pipeline inventory examines the probability that sometime during the program so many vehicles will be lost that there will not be sufficient vehicles to sustain the launch rate. The maximum number of vehicles which can be lost at any particular time in the program without disturbing the scheduled launch rate is the difference between the initial quantity of vehicles and the number of vehicles required for the pipeline at that particular time. As the turnaround time decreases the number of pipeline vehicles decreases and the number of vehicles which can be lost without disturbing the launch schedule increases. The probability of falling below pipeline requirements is calculated at the critical flights. A critical flight is the last flight before the number of pipeline vehicles decreases by one. There may be several critical flights during a program as indicated in figure 9-3. At each critical flight, the allowable number of vehicles is calculated and also the probability of exceeding this allowable number. If the probability exceeds 10%, an additional vehicle is added to the total pipeline inventory.

9.2.2.3 ADDITIONAL VEHICLES - At this stage in the logic, the only remaining problem is whether the total number of flights required for the program can be achieved by the current inventory total. The current inventory consists of two vehicle types: 1) NA vehicles which will be lost sometime during the program and 2) all the remaining vehicles which will complete the entire program. If the non-abort vehicles had unlimited use capability, i.e. no design life or limit, then the current inventory would always be capable of achieving the total launches required of any program. If the nonabort vehicles are limited, however, the current inventory may not be capable of the total flights

FIGURE 9-2

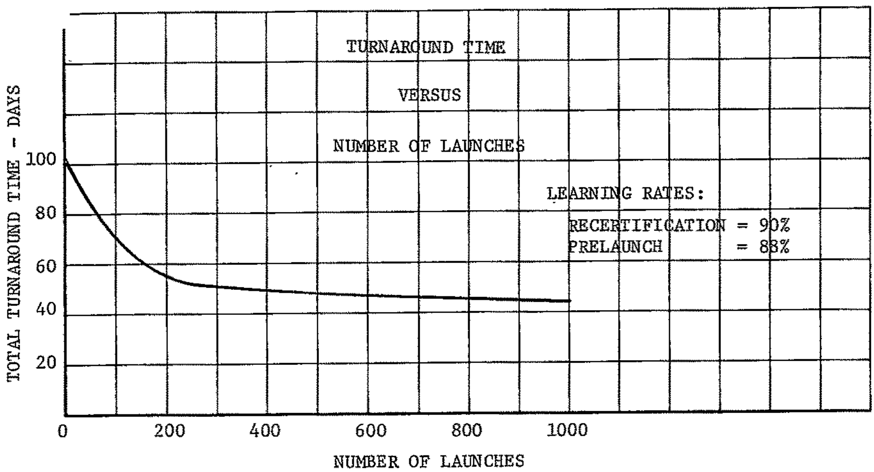
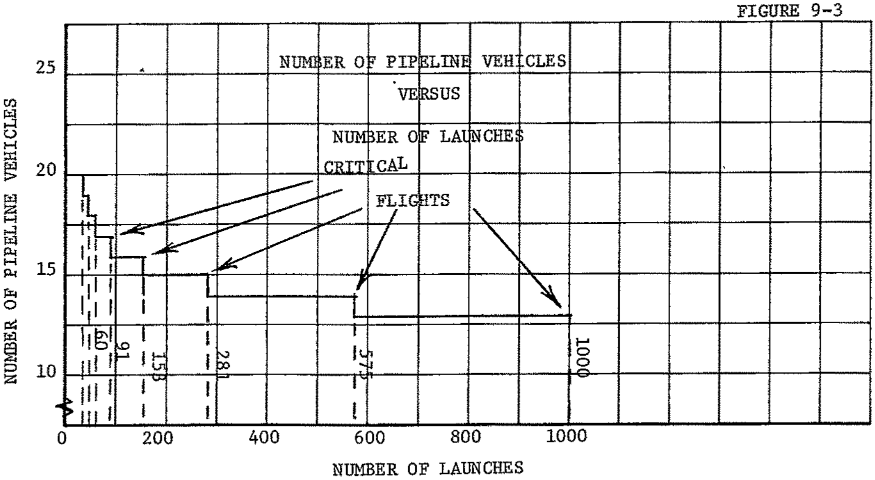


FIGURE 9-3



required for the program. The total flights which the current inventory can be used for is given by

9.2-5 TNOFLTS = (NL+NR) x (DL) + (NA) x (NOFLTS)

TNOFLTS = total flights from current inventory

DL = design life of each vehicle (number of launches)

NOFLTS = expected number of flights from each NA vehicle

If TNOFLTS exceeds or equals NOAL, the current inventory is sufficient and no additional vehicles are required. If TNOFLTS is less than NOAL, one vehicle is added to the current inventory and a new TNOFLTS is calculated and compared with NOAL. Additional vehicles are added until TNOFLTS exceeds or equals NOAL.

The number of flights, NOFLTS, which each NA vehicle is used for is calculated assuming the vehicle aborts are uniformly distributed between launch one and average number of flights expected from all vehicles. The average number of flights is obtained from

9.2-6 $\text{NOFV} = \begin{cases} \text{DL} & \text{if } \text{NOAL}/(\text{NA+NL}) \geq \text{DL} \\ \text{NOAL}/(\text{NA+NL}) & \text{otherwise} \end{cases}$

Equation 9.2-6 insures that the average number of flights never exceeds the design life of the vehicles. If after the design life limitations have been satisfied, the current inventory is less than two vehicles, then an additional vehicle is added to bring the total inventory up to a minimum of two vehicles.

9.3 CARGO/PROPULSION SECTION INVENTORY - In the OCPDM study, two configurations, A and B, use an expendable cargo/propulsion module. Since a new module is required for every launch, the total inventory of modules must equal the total number of launches.

In the remaining configurations C, D, E and F, the cargo/propulsion section is integral with the crew section and consequently is part of the reentry vehicle. For these configurations the cargo/propulsion section inventory is equal to the number of reentry vehicles required.

9.4 LAUNCH VEHICLE INVENTORY - The OCPDM study includes six reuse concepts ranging from an all expendable modular vehicle to an all integral vehicle with a reusable first stage. The reuse concepts A, B and C use an expendable first and second stage launch vehicle. Concepts D and E use an expendable first stage and concept F uses a reusable first stage.

For concepts A through E, which use an expendable launch vehicle, the number of launch vehicles is set equal to the number of attempted launches. For the reusable first stage of concept F, five vehicles are assumed for the development phase and these five launch vehicles are considered adequate for the investment phase for any annual launch rate up to 30/year.

9.5 SIGNIFICANT VARIABLE NAMES - INVENTORY SUBROUTINE

ABAR - true annual launch rate

ALPHA - confidence level on the number of attempted launches

ALPHA2 - confidence level on the number of abort vehicles

CRITT - critical turnaround times used to calculate the number of pipeline vehicles

DL - design life of the entry vehicles (user input)

EM - expected value of the mean from a uniform destruction with range NOFV.

EXX - learning rate exponent on the recertification time (set at -.152, i.e. 90%)

FLTN - flight number counter used in calculating critical flights

LONOV - number of vehicles required at the minimum turnaround time

LR - annual launch rate (user input)

MAXVEH - number of vehicles required at the maximum pipeline time

MAXW - vehicle counter used in calculating the critical flights

MNTT - minimum turnaround time in days

MR - total mission reliability

MT - total mission time in days

MXTT - maximum turnaround time in days

NA - quantity of abort vehicles

NL - total quantity of pipeline vehicles

NOAL - total number of attempted launches

NOFLTS - expected number of flights from each abort vehicle

NOFV - average number of flights required from the entire inventory to complete the required number of attempted launches

NOSM - number of successful missions

NOV - total current inventory

NOV(2) - entry vehicle

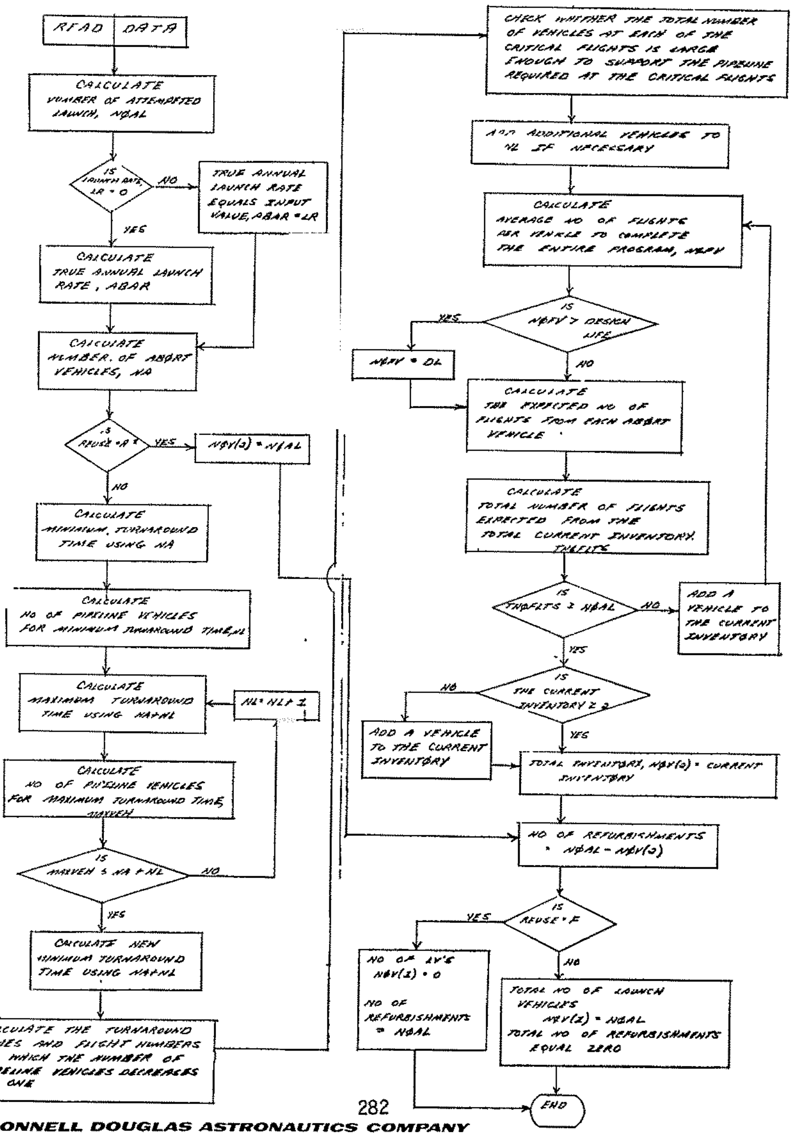
NOV(1) - launch vehicle

NR	-	quantity of additional vehicles
NRFB	-	number of refurbishments for the entry vehicles
NRFB1	-	number of refurbishments for the launch vehicles
OST	-	orbit stay time in days (user input)
PL	-	operational program length in years (user input)
POSR	-	launch to launch reliability of each entry vehicle (user input)
REUSE	-	reuse concept of the entry vehicle (user input)
RT	-	return time from orbit to recovery in hours
SD	-	standard deviation of the estimated mean of a uniform distribution with range NOFV after NA observations
TCD	-	total calendar days for recertification (first unit)
TNOFLTS	-	total number of flights possible from the current inventory
TPL	-	total days for prelaunch activities (first unit)
TT	-	transportation time in days
VEHMAX	-	number of pipeline vehicles required at the maximum turnaround time

9.6 INVENTORY MODEL LOGIC FLOW - The inventory model logic flow is illustrated in Figure 9-4.

INVENTØRY SUBRØUTINE LØGIC FLØW

FIGURE 9-4



10.0 SUBSYSTEM SELECTION MODEL

10.1 INTRODUCTION - One approach to subsystem optimization would be to to determine program cost for every possible combination of subsystem alternates and choose the combination that gives the lowest cost. Applying this approach to the vehicle configurations in this study would result in up to 4,400,000 combinations, depending upon which vehicle configuration is under investigation. Each combination must be sized in a sizing program. Since this is impractical, another approach is required. The following paragraphs define an approach that requires only determining the program cost for each subsystem alternate individually plus a few checks to determine cost sensitivity to weight changes.

The subsystems that are to be varied to give the combination that results in the lowest program cost in this study are: primary structure, thermal protection system, upper stage boost propulsion system, orbit maneuver propulsion system, electrical power supply, hydraulic system power supply, environmental control system, guidance and control system, and telecommunications. Each of these subsystems has from two to twelve alternate configurations. An investigation of the interactions between these subsystems, within the groundrules of this study, reveals that between some there is no interaction, between some there is a one way interaction, and between others there is a two way interaction. These interactions are shown in Figure 10-1.

If the two way interactions between subsystems are evaluated first and the lowest cost combination defined, then the block of subsystems that are mutually independent and have only a one way interaction with the remainder of the subsystems can be evaluated one at a time to chose the alternate of each subsystem that results in the lowest program cost.

10.2 SIGNIFICANT VARIABLE NAMES - The names of descriptions of the significant variables in this model are given in Table 10-1. These are included in the list in Paragraph 5.2 and are repeated here for convenience.

FIGURE 10-1

SUBSYSTEM INTERACTION

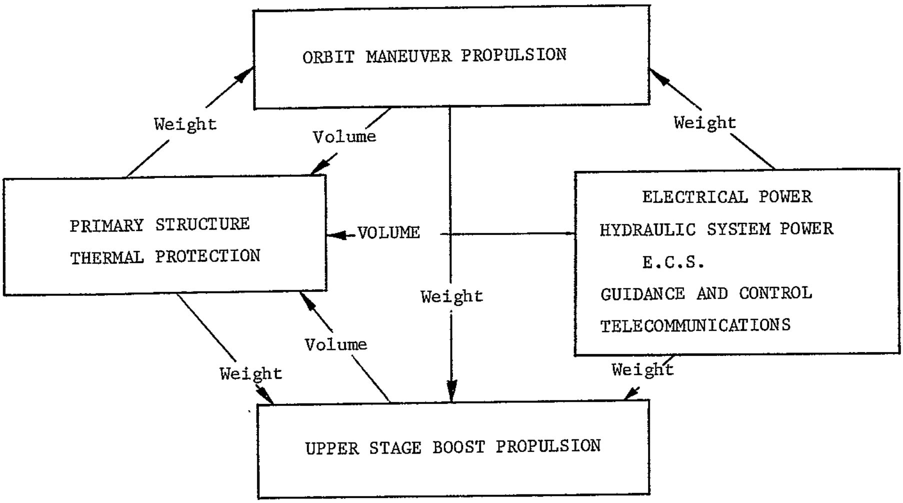


TABLE 10-1

SIGNIFICANT VARIABLES - SUBSYSTEM SELECTION MODEL

ALT(I,J) -	Jth alternate for the Ith subsystem
BALT(I) -	Baseline alternate for the Ith subsystem
C(I) -	Total program cost in millions of dollars
ISP -	Specific impulse of the propellant in feet/second
MINC(I) -	Total program cost in millions of dollars for the minimum cost alternate of the Ith subsystem
NISP -	Specific impulse of the propellant in feet/second
OMB -	Orbit maneuver baseline alternate
PC(I, J) -	Total program cost in millions of dollars for the Jth alternate of the Ith subsystem
R(I) -	Slope of the program cost versus orbit manevuer or upper stage boost weight
SS -	Subsystem indicator
WTOM(I, J) -	Weight of the orbit maenvuer system in pounds for the Jth alternate of the Ith subsystem
WTUS(I,J) -	Weight of the upper stage propulsion in pounds for the Jth alternate of the Ith subsystem

10.3 DESCRIPTION OF LOGIC - The detail logic for the subsystem selection model is shown in Figure 10-2.

At the start of the selection procedure, one alternate of each subsystem is designated as the baseline. These baselines are used in all subsequent steps until replaced as a result of an optimization procedure. Since the primary structure of the entry vehicle and the thermal protection system are independent of each other and since they have the same interactions with the other subsystems, these are evaluated first.

Each of the alternates for the entry vehicle primary structure (STEV) are evaluated one at a time. For each, the vehicle is sized, and the program cost(PC), the weight of the orbit maneuver system (WTOM), and the weight of the upper stage boost system (WTUS) are determined. The structural alternate that results in the lowest program cost is then defined as the new baseline.

Next, it is determined if there are any of the STEV alternates that are lighter than the present baseline. Since variations in structural concepts affect the weight of the structure only and not the volume it occupies, the weight of the primary structure is reflected in the weight of the orbit maneuver system and the upper stage boost system. A lighter structural alternate means a reduction in propulsion system weight and size and a resulting decrease in vehicle size. Use of a lighter structure may result in a lower program cost if the baseline for a propulsion system changes during the subsequent propulsion system optimization. If there are lighter alternates, then $\partial PC/WTOM$, the increase in program cost per pound of reduction in orbit maneuver system weight (which would result from the lighter structural alternate), is determined for each. Figure 10-3 shows how this would look if plotted. The two alternates that are lighter than the baseline represent potential weight savings and thus cost savings in the propulsion systems at the expense of increased structural costs. The second subroutine repeats the above except with respect to the weight of the upper stage boost if the vehicle is integral.

This procedure is then repeated for the thermal protection system (TP) and the orbital maenvuer propulsion system (OM) alternates (and for the mission module structural (STMM) alternates if the spacecraft is modular). For each of these the $\partial PC/\partial WTOM$ and/or $\partial PC/\partial WTUS$ is determined for any alternate that is lighter than the baseline.

SUBSYSTEM SELECTION LOGIC DIAGRAM

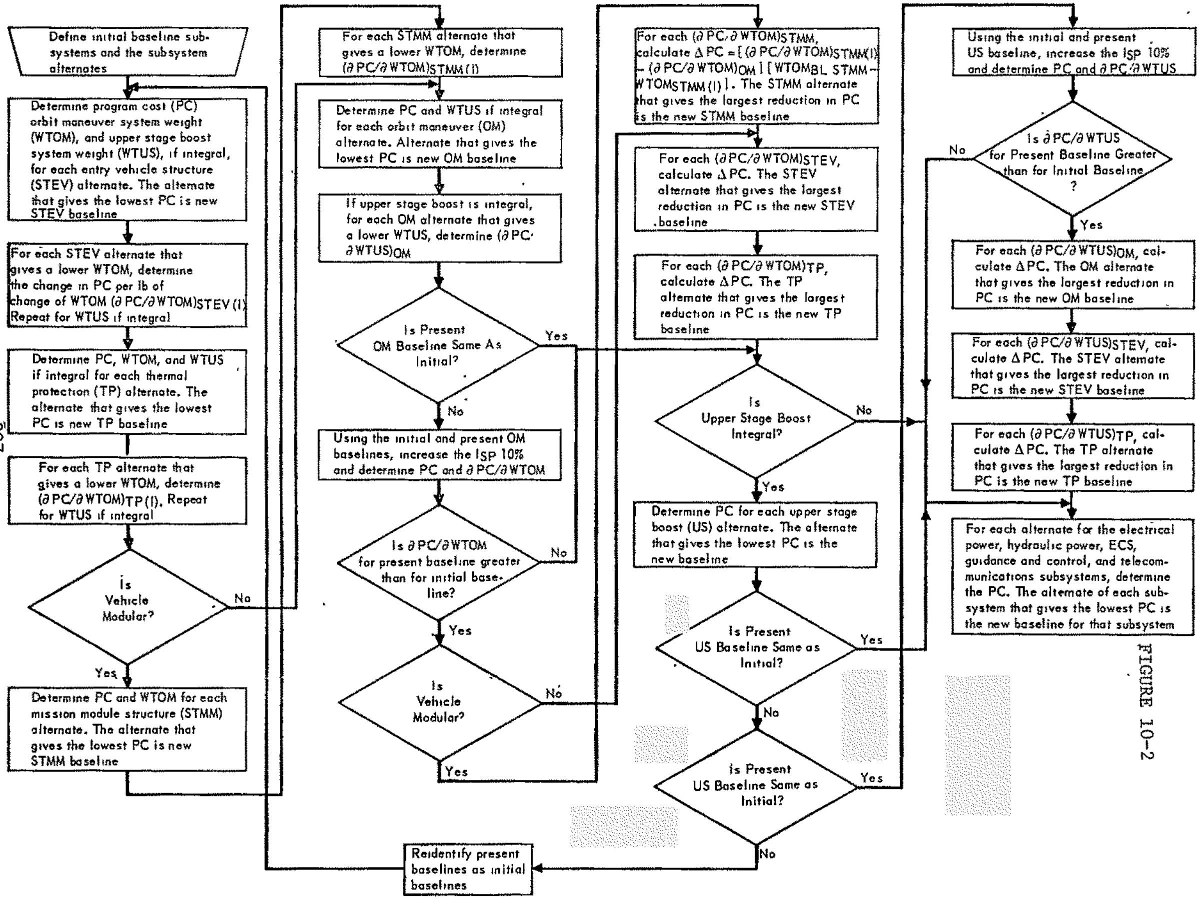
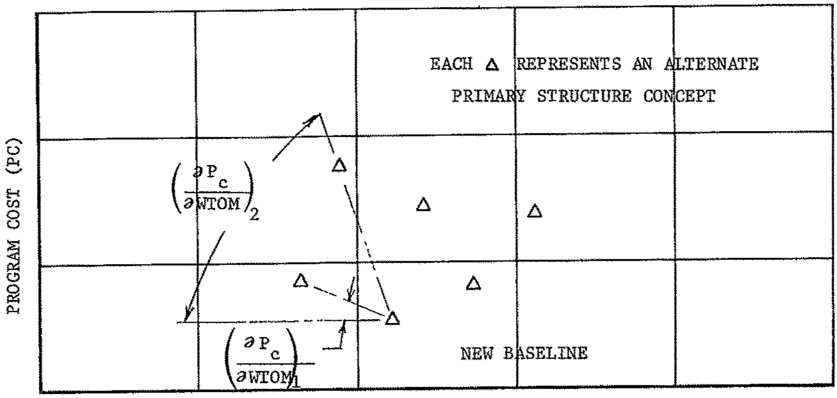


FIGURE 10-2

FIGURE 10-3

DETERMINATION OF ∂PC/∂WTOM



At this point, a check is made to see if the orbit maneuver system baseline has changed from the one used up to this time. If it has not, then the choice of baselines previously made (i.e., structure and thermal protection) are still valid. If there has been a change in the orbit maneuver baseline, it is possible that the choice of the structural and thermal protection baselines can be improved by going to a lighter alternate if the program cost decreases more rapdily with a reduction in orbit maneuver weight for the new baseline than with the initial baseline.

To determine if a change in structural or thermal protection baselines is warranted, the specific impulse for both the initial and new baseline orbital maneuver systems is increased. This artifically reduces the weight to the orbital maneuver system without changing any other subsystem except those affected by the size of the orbital maneuver system and gives the reduction in program cost with a decrease in orbit maneuver system size. If the reduction (decrease in program cost/weight saving in orbit maenuver system) for the new baseline is equal to or less than the reduction for the initial baseline then there is no need to reevaluate the structural and thermal protection baselines. If the reduction is greater, then the previously chosen structural and thermal protection baselines may not be optimum if there were lighter alternates.

It is therefore necessary to check for each subsystem that has been previously evaluated. This is done by comparing, for each alterante that is lighter than the present baseline, the increase in program cost that results in using a lighter but more expensive alternate to the decrease in program cost that results from a lighter orbit maneuver system.

For each mission module primary structural (STMM) alternate that is lighter than the baseline as determined earlier, a calculation is made. It is determined whether the cost reduction resulting from the decrease in orbit maneuver weight that resulted from a lighter structure more than offsets the increased cost of the lighter structure. If any of the lighter alternates show a reduction in program cost, the one that gives the greatest is selected as the new baseline in the following analyses.

The above procedure is repeated for the entry vehicle structure and the thermal protection system to determine if a cost reduction results from using a lighter alternate.

If the upper stage boost propulsion system is integral with the entry vehicle, a procedure very similar to that used for the orbital maneuver propulsion system is used to pick a baseline and then to determine if the previous choice of baselines is still valid. The only difference here is there is no mission module for an integral vehicle but the orbit maneuver system baseline must be checked.

If both the orbit maneuver and upper stage boost system baselines change during the selection procedure, the baselines determined up to this point are designated initial baselines and the selection procedure starts at the beginning again. This is to give input data closer to the final answer so there will be less of a chance for error in the final results.

Determine Baseline for other S/S - With the exception of the hydrauli system, the remaining systems (Environmental Control, Electrical power supply, Guidance & Navigation, and Telecommunications) are not affected by vehicle size or weight and do not affect each other within the limits of the OCPDM study.. Because of this, each alternate of each of these subsystems can be checked one at a time and the one that gives the lowest program cost is then chosen as the baseline.

The hydraulic system power requirements are very much dependent upon vehicle size and the volume requirements for the hydraulic system power supply are dependent both upon power requirements and the choice of alternates. In this study there are only two alternates, one of which will probably never be competitive because of its extremely high weight. Therefore, the interaction is not considered.

10.4 LIMITATIONS AND ASSUMPTIONS - The statement that there is no interaction between the Guidance and Navigation, Telecommunications and Environmental Control and the Electrical Power System is true only if the power requirements for the various alternates are approximately the same. In this study, the difference in power requirements is small for the various alternates for a given vehicle configuration and crew size. Since these are held constant during subsystem selection the logic is valid.

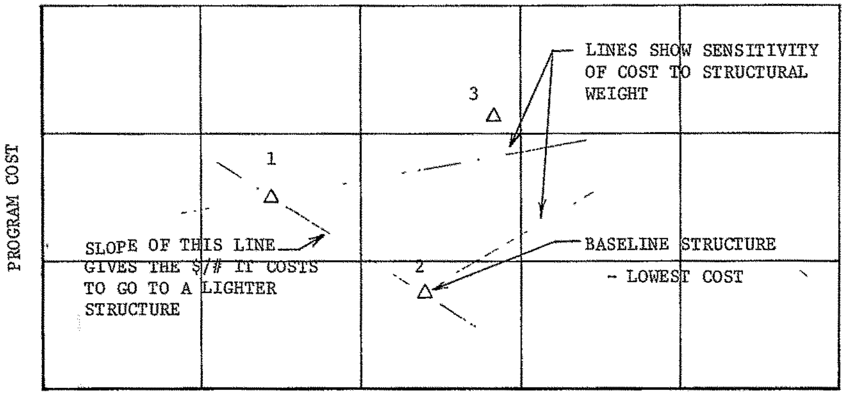
When checking lighter but more expensive alternates against the cost saving of lighter weight propulsion systems, the $/# saving for the propulsion system is based on baseline subsystems. It is possible that the savings would not be as great using a non baseline system. Figure 10-4 shows graphically what the logic does when considering lighter weight systems. If, as shown in the figure, the program cost is not as sensitive to variations in weight of the lighter weight alternate, then the sensitivity of cost to OM weight will be less than the program calculates. This is shown by the dotted line in the lower figure and results in less cost saving than calculated. The logic could be changed to check sensitivity using the lighter weight alternates but it would have to be done for each lighter alternate for each subsystem checked and could increase computation time considerably. It is felt that the improvement is second order and not worth the increased complexity.

This logic assumes that, when selecting the optimum alternate for electrical power, hydraulic system power supply, ECS, Guidance and Control, and Telecommunications that the condition shown in Figure 10-5 does not exist or that the cost difference is so small as to be negligible. In Figure 10-5, alternate #1 was chosen as the baseline because it gave a lower program cost.

At the time of choosing the optimum OM system, the above listed subsystems had not been optimized but were using the initial baselines. If, when optimizing these subsystems the weight of the OM decreases to the point where it is to the left of the intersection of the sensitivity lines, then the optimum orbit maneuver system has not been chosen. It is assumed that any error due to this occurance is second order and negligible.

FIGURE 10-4

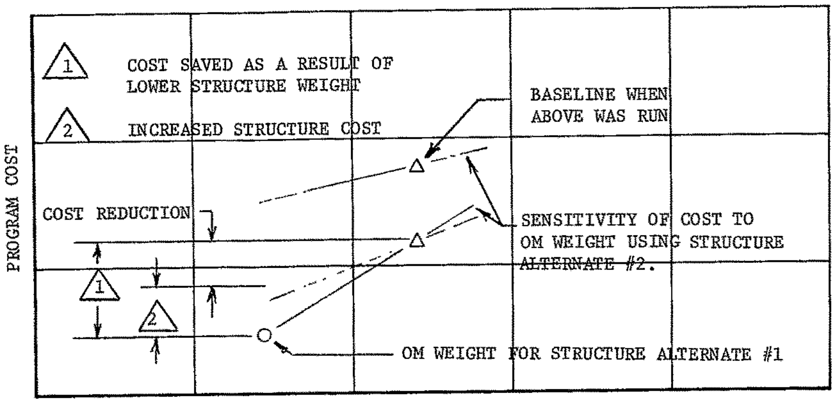
LIMITATION FROM DIFFERENCES IN COST SENSITIVITY



WEIGHT OF ORBIT MANEUVER SYSTEM AS STRUCTURAL ALTERNATES ARE VARIED

FIGURE 10-4
(CONTINUED)

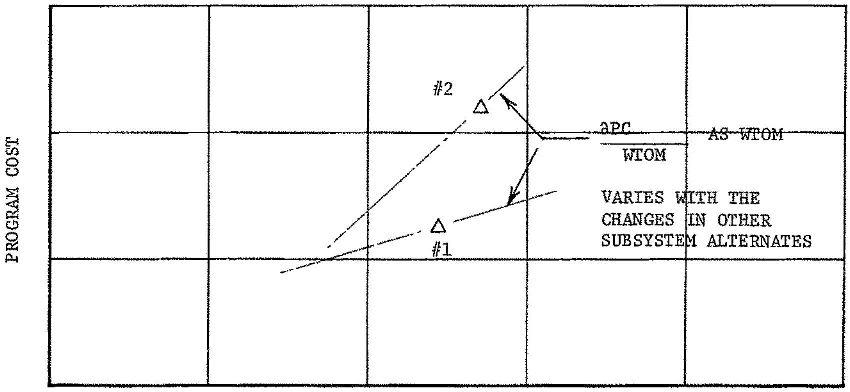
LIMITATION FROM DIFFERENCES IN COST SENSITIVITY



ORBIT MANEUVER SYSTEM WEIGHT AS ORBIT MANEUVER ALTERNATES ARE VARIED

FIGURE 10-5

LIMITATIONS FROM DIFFERENCES IN COST SENSITIVITY OF SUBSYSTEM ALTERNATES



ORBIT MANEUVER SYSTEM WEIGHT

RELIABILITY OPTIMIZATION MODEL

11.0 RELIABILITY OPTIMIZATION MODEL - The object of the reliability optimization model is to minimize the development and investment costs for individual subsystems by reallocating the subsystem reliabilities. Several side conditions must also be satisfied

a) An overall subsystem reliability must be met.

b) The individual subsystems may have upper and/or lower limits on their reliabilities.

11.1 MODEL THEORY

11.1.1 COST RELIABILITY RELATIONSHIP - In order to reallocate the reliabilities using cost as the driving parameter, it was necessary to postulate possible functional relationships between reliability and cost. Obviously, several properties can easily be assumed:

a) The reliability will only be defined between zero and unity.

b) The cost must always be positive.

c) The cost increases with increasing reliability.

d) The cost should become prohibitive as the reliability gets very near one.

Several functions can be defined which will satisfy the above assumptions. The OCPDM model uses the form:

11.1-1 $C = C_o[1 + b\ln(\frac{1-R}{1-R_o})]\quad 0 \leq R_o,\ R \leq 1,\ b \leq 0.$

where:

C = cost of the system at R reliability

C_o = Reference cost related to the reliability R.

b = Equation parameter

R = Independent variable, reliability

R_o = Reference (baseline) reliability

The reference cost used in each equation will be generated by the spacecraft cost model and input into the reliability model. For each subsystem there will be two equations:

a) development cost versus reliability

b) investment cost versus reliability

The reference cost for a) will be the total design and development cost generated by the cost model. The reference cost for b) will be the total investment cost for that subsystem. It is assumed that all costs from the cost model are related to the reference reliability for each subsystem. These reliabilities are permanent data in the program and are shown in Table 11-1.

The "b" parameter shown in equation 11.1-1 controls the cost increase with reliability increase. To fully illustrate the significance of the "b" parameter, the equation can be rewritten with the following substitutions:

$$U = 1 - R$$

U - Unreliability

$$b = \frac{B}{100 \ln(1/2)}, \quad \ln(1/2) = -.69315$$

The equation now becomes

11.1-2 $$C = C_o[1 + \frac{B}{100 \ln(1/2)} \ln (\frac{U}{U_o})]$$

The new parameter "B" is a function of "b" only, but in the form of 11.1-2 the "B" parameter has a more direct physical definition. In equation 11.1-2, "B" is the percent of the reference cost, C_o, which is added to the total cost every time the unreliability is halved, e.g. of U = .16, C_o = 10 million dollars and B is set equal to 20, then

from U =	to U =	ΔCost
.16	.08	2M
.08	.04	2M
.04	.02	2M

and the list could continue on and on. The important feature is that although the decrease in absolute unreliability becomes smaller, the cost for the decrease remains constant.

11.1.2 COST MINIMIZATION - Once the cost-reliability relationship has been defined the problem can be expressed more compactly. The total cost of N subsystems is:

11.1-3 $$C = \sum_{L=1}^{N} C_{D_i} + \sum_{L=1}^{N} C_{I_i}$$

where:

C = total cost

C_{D_i} = development cost for subsystem i

C_{I_i} = investment cost for subsystem i

TABLE 11-1

RELIABILITY APPORTIONMENT SUMMARY

LOGISTIC MISSION

Subsystems	Modular		Partial Integral		Integral with Tip Tanks		Fully Integral	
	Ballistic	Lifting Body	Ballistic	Lifting Body	Ballistic	Lifting Body	Ballistic	Lifting Body
a) Upper Stage Propulsion	.9750	.9750	.9750	.9750	.9750	.9750	.9750	.9750
b) ECS	.9975	.9975	.9975	.9975	.9975	.9975	.9975	.9975
c) Communications	.9982	.9982	.9982	.9982	.9982	.9982	.9982	.9982
d) Power (Electrical)	.9971	.9978	.9971	.9978	.9971	.9978	.9971	.9978
e) G&C, Radar (Electronics)	.9967	.9967	.9967	.9967	.9967	.9967	.9967	.9967
f) Reentry Control Propulsion	.9965	.9966	.9965	.9966	.9965	.9966	.9965	.9966
g) Maneuver and Orbital Attitude Control	.9968	.9968	.9968	.9968	.9968	.9968	.9968	.9968
h) Primary Structure	.99934	.99942	.99934	.99942	.99934	.99942	.99934	.99942
i) Thermal Protection	.99956	.99968	.99956	.99968	.99956	.99968	.99956	.99968
j) Vertical Landing and Recovery	.9974	N/A	.9974	N/A	.9974	N/A	.9974	N/A
k) Instrumentation	.9974	.9943	.9973	.9973	.9973	.9973	.9973	.9973
l) Sequentials, Pyros, Docking, and Retro	.9986	.9986	.9986	.9986	.9986	.9986	.9986	.9986
m) Aerodynamic Control	N/A	.9985	N/A	.9985	N/A	.9985	N/A	.9985
n) Horizontal Landing and Recovery	N/A	.9979	N/A	.9979	N/A	.9979	N/A	.9979
R. Total	.9500	.9500						

* For design cases where the maneuver subsystem and the orbital attitude control subsystem are separate subsystems, the apportionment is: Maneuver Subsystem = .9987
Orbit Attitude Control = .9981

Escape system apportionments are not applicable for mission success.

The function in 11.1-3 must be minimized with the side conditions that

11.1-4
$$R = \prod_{=1}^{N} R_i \geq R_o$$

where:

R = overall reliability goal

R_i = reliability for subsystem i

and

11.1-5
$$L_i \leq R_i \leq U_i \qquad i = 1, 2, \ldots, N$$

where:

L_i = lower reliability limit for subsystem i

U_i = upper reliability limit for subsystem i

The inequality in 11.1-4 states that the product of all of subsystem reliabilities must be equal to or greater than the overall reliability goal. The unequality in 11.1-5 states that each subsystem reliability must be within the reliability limits set for it. Both the overall subsystem reliability and the upper and lower reliabilities for each subsystem are user inputs as defined in Section 5.3.5 of this book.

If the restrictions in 11.1-5 were removed, the minimization of equation 11.1-3 with the side condition of 11.1-4 could be done directly using Lagrangean multipliers. Briefly, this method would find the N partial derivatives of the total cost versus the unreliability of each subsystem from 11.1-3 and set them equal to zero, i.e.,

$$\frac{\partial C}{\partial U_i} = \frac{-C_i^D \, b_i^D}{U_i} + \frac{C_i^I \, b_i^I}{U_i} = 0$$

There are two terms for each partial derivative because there are two equations for each subsystem. The N partial derivatives set all the slopes equal to each other and for a given overall unreliability this minimizes 11.1-3. The side condition 11.1-4 is used to find which overall unrealiability minimizes the cost. The answer is physically obvious. Since cost is a monotonically increasing function of unreliability, the lowest total cost is obtained with the lowest overall reliability possible. In this case, that overall reliability is the user input. This method requires that the individual reliabilities are not constrained. Because of 11.1-4, an iterative method of finding the optimum set of reliabilities is used.

The basic concept of the iterative process is the fact that if more reliability is required, then the reliability should be added to the subsystem(s) which have the smallest cost increase with reliability increase, i.e., the smallest or lowest slope. The iterative solution begins by assigning each subsystem its lower reliability limit. This set of reliabilities is the cheapest combination possible. If the product of the lower limits already exceeds the overall reliability goal, then no iteration is necessary and the set of lower limits is the solution set. In most cases, however, the set of lower limits will be below the overall reliability goal and the subsystem reliabilities must be increased.

Before the iterative increase in reliability begins, the program calculates the slopes for every upper and lower reliability limit for each subsystem and arranges these slopes in ascending order. These slopes serve as "stepping stones", i.e., the program knows that while the iterative slope is between any two stepping stones, no subsystem will be beginning or ending its gain in reliability. Having these steps, the program finds the lowest slope and the subsystem(s) associated with it. It immediately raises the slope of each of these subsystems to the next step. The increase in slope increases the subsystem reliability and a new product of the subsystem reliabilities is calculated and compared with the overall reliability goal. If the goal has been exceeded, the program knows it has bounded the optimum slope and proceeds to iterate to the final answer. If the reliability product is still short of the reliability goal, the program prepares to jump to the next step. It first determines which subsystems are still growing in reliability and then iterates once again. Here the slope logic is repeated as described above and the program continues stepping until the reliability goal is exceeded and then iterates to the final answer. To assure that a solution does exist, the program also calculates the product of all the upper reliability limits initially and compares this product with the reliabiltiy goal. If the product of all the upper limits is still less than the goal, then no solution is possible.

11.2 <u>OCPDM MODEL</u> - The reliability optimization model uses cost as the driving parameter and accordingly, the OCPDM model uses those subsystems which have the largest effect on the total program cost. The subsystems used and the reference reliabilities are shown in Table 11-2. This is not a complete list of all the subsystems used in the total reliability allocation. Table 11-1 has the entire subsystem reliability allocation.

For each subsystem shown in Table 11-2, the model has both the total development and investment cost from the spacecraft cost model. Using the reference reliabilities shown and a set of "B" parameters, the model will reallocate the reliabilities to minimize the total program cost.

TABLE 11-2

RELIABILTY OPTIMIZATION SUBSYSTEMS

B	SUBSYSTEMS	BASELINE RELIABILITIES	
		BALLISTIC	LIFTING BODY
1	Thermal/Structure	.9989	.9991
2	Power	.9971	.9963
3	ECS	.9975	.9975
	Avoinics		
4	Guidance & Control	.9967	.9967
5	Telecommunications	.9982	.9982
	Propulsion		
6	EACS	.9965	.9966
7	Vernier*	.9987	.9987
8	Main Maneuver	.9981	.9981
9	Upper Stage	.975	.975

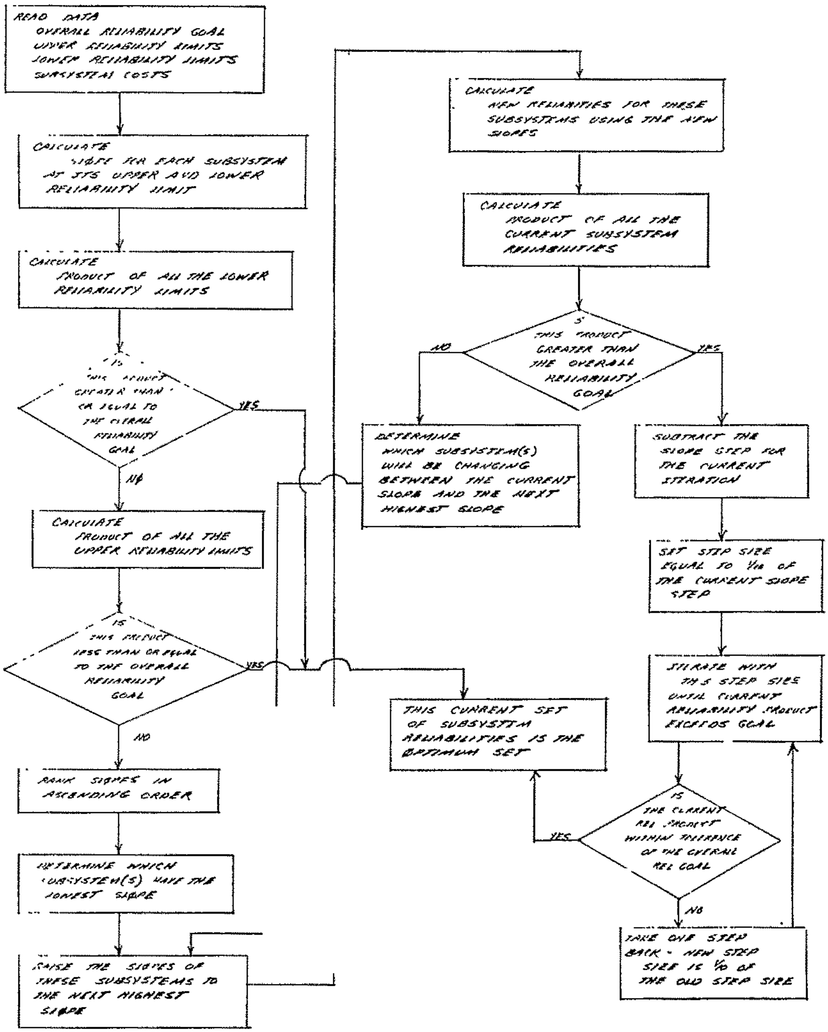
* For design cases where the main maneuver functions are performed by the vernier maneuver system, the reliability of the vernier system is .9968.

The "B" parameters are not regular user inputs to the OCPDM program. The values for the "B" parameters are read in as permanent data for every program run. The program user can over ride one or all of these "B" values by inputing new values with his regular input data. The B array has nine slots corresponding to the nine subsystems shown in Table 11-2.

11.3 RELIABILITY OPTIMIZATION LOGIC FLOW - The reliability optimization logic flow is illustrated in Figure 11-1.

RELIABILITY COST OPTIMIZATION
LOGIC FLOW

FIGURE 11-1



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12.0 CARGO SIZE OPTIMIZATION - There are three modes available to the OCPDM program user for determining the spacecraft size

a) fixed cargo weight/launch
b) fixed spacecraft weight
c) optimum cargo weight/launch (golden rule search)

The data input requirements are explained in Section 5.4.2 of this book.

12.1 FIXED CARGO WEIGHT/LAUNCH - The program user can either input the cargo weight/launch directly or else define the value by inputing the other parameters for the equation:

$$\text{(cargo wt/launch)} = \frac{\text{(total cargo wt)}}{\text{(annual launch rate)} \times \text{(program length)}}$$

Once the cargo weight/launch is known, this value is input into the sizing model which determines the total spacecraft size and weight with the cargo capability desired. For this mode a rubber launch vehicle is assumed. After the total spacecraft weight has been determined, the launch vehicle throw weight capability is calculated. For two stage launch vehicles the total spacecraft effective weight is adjusted to a due east ETR launch. This adjusted weight is used in the CER's for the launch vehicle. For single stage launch vehicles, the total spacecraft gross weight is adjusted. The spacecraft effective weight is the weight which a two stage launch vehicle must be capable of injecting into orbit. The effective weight is less than the gross weight because of the hardware, primarily the abort tower, which is not carried all the way to orbit. The throw weight capability for the first stage only launch vehicles is the gross weight since the first stage must be capable of lifting the entire spacecraft and does not benefit from any jettisoned hardware. The adjustment factors for the launch vehicle throw weight capability from 50°, 70° or 90° orbit inclination to a due east launch from ETR are shown in Table 5-4 in Section 5.4.2 of this book.

12.2 FIXED SPACECRAFT WEIGHT - This particular mode can be used when the program user is required to use a specific launch vehicle with fixed throw weight capability. The data inputs for this mode are shown in Table 5-2 of Section 5.4.2 of this book. The program user inputs the launch vehicle throw weight capability for a due east ETR launch. The program adjusts this capability for the launch site and orbit inclination being used. This adjusted weight is input into the sizing model as the spacecraft effective weight.

The sizing model designs the entire spacecraft to meet this effective launch weight.

The program user must take care that he inputs the correct launch vehicle throw weight capability. For example, if the launch vehicle to be used can lift 10,000 pounds into a 50° orbit from ETR, then the user must input an adjusted capability for that launch vehicle for a due east launch from ETR. The adjusted weight is obtained from

$$LVTW_i = P_i \; LVTWE$$

where:

$LVTW_i$ = launch vehicle throw weight capability for an i orbit inclination

P_i = degradation factor for an i orbit inclination

LVTWE = throw weight capability for a due east launch from ETR

therefore

if $LVTW_{50°} = 10{,}000$

$P_{50°} = .9488$

then $LVTWE = 10539$

12.3 OPTIMUM CARGO WEIGHT/LAUNCH - The search for an optimum cargo weight/launch is really a series of fixed cargo weight/launch runs. The program user inputs the upper and lower limits of the initial search range and the width of the final range. The optimum cargo weight/launch is defined as the cargo weight/launch which minimizes the total program cost. All parameters are held constant during the search except cargo weight/launch. The user inputs for a golden rule search are shown in Table 5-3 in Section 5.4.2 of this book.

12.4 GOLDEN RULE SEARCH - The process described above is a one-dimensional search, i.e., one parameter is to be optimized, cargo weight/launch. The most effective technique for locating the minimum of a unimodal one-dimensional function is a Fibonacci search. To use this technique, the final accuracy must be specified initially and this represents an often serious problem since the final accuracy is somewhat related to the function being examined.

This difficulty can be overcome with little loss in search efficiency by using an alternate technique based on the so-called Golden Section or Golden Rule Search.

The golden rule search begins by evaluating the program costs at each end of the initial search range and at $G = 2/(1 + \sqrt{5})$ of the range from both of the end points. These first four search points are shown in Figure 12-1. The order in which these first four points are calculated is not important and in Figure 12-1, the search points are numbered in the same order as the program would calculate them. The boundary point furthest from the lowest cost point is now discarded. In Figure 12-1, the number 3 point is the lowest and consequently the left boundary point is discarded. The other three points are retained and the search continues in the interval bounded by the number 2 and 4 points. The search interval has been diminished in size by G^2 since $1-G = G^2$. The new point is evaluated at a distance G^2Lo from the new left boundary. The new set of four points; 2, 3, 4 and 5; have the identical geometric relationship to one another as the first four points had. The new interval is approximately 39% shorter than the original interval. The search logic now repeats itself. The boundary point furthest from the lowest cost point is discarded. In Figure 12-1 this would be point 4. The new interval will be bounded by points 2 and 5. The new point will be added at a distance G^3Lo from the right boundary, between points 2 and 3. This new interval will be 39% smaller than the previous interval. The unique feature of this search technique is the constant reduction in the search interval size. After Q evaluations, the minimum cost point will be known to within R of the original search range where:

12.4-1 $$R = G^{Q-3}$$

This function is shown in Figure 12-2. The program user can control the number of iterations through the sizing and cost models. Table 12-1 indicates the maximum ratio of initial range to final range that can be obtained with a certain number of iterations.

FIGURE 12-1

GOLDEN RULE SEARCH

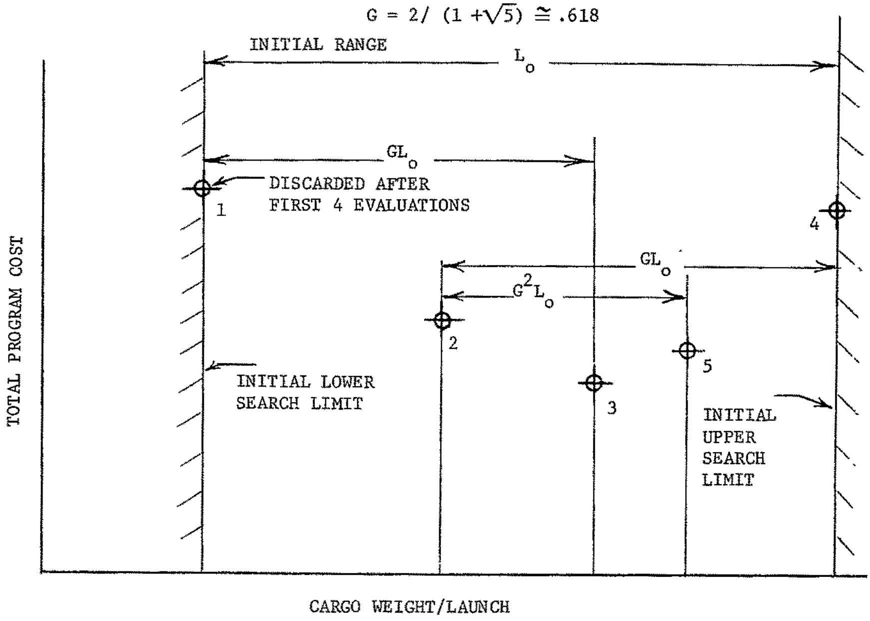


FIGURE 12-2

INTERVAL LENGTH VERSUS NUMBER OF SEARCH CYCLES

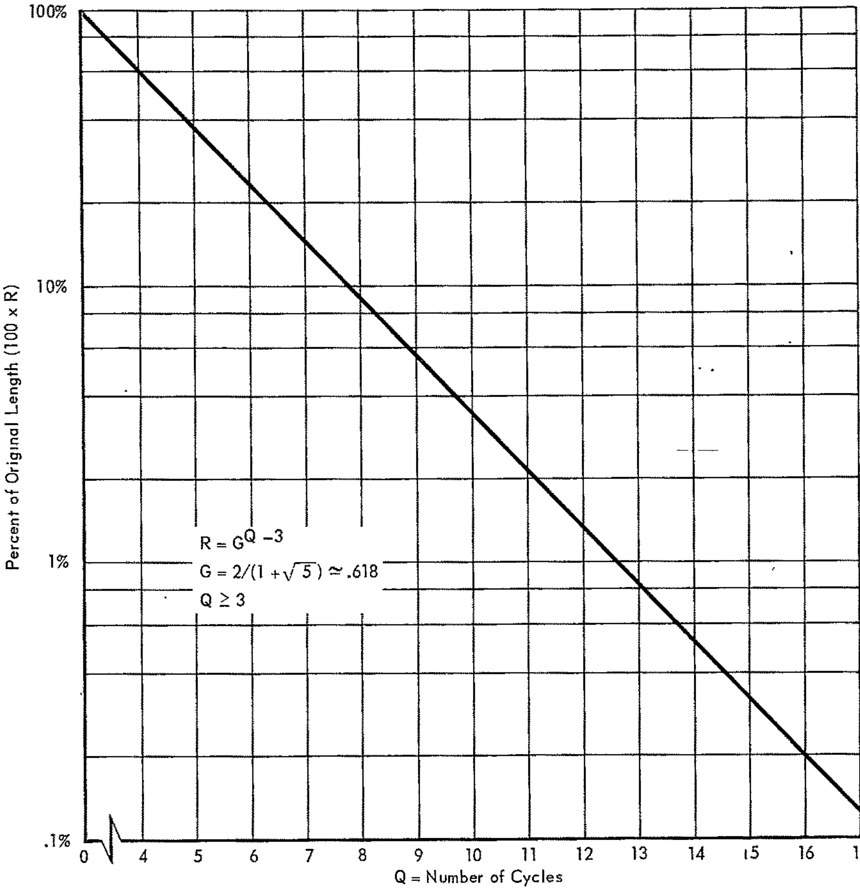


TABLE 12-1

GOLDEN RULE ITERATIONS

ITERATIONS	REDUCTION RATIO*	PER CENT OF ORIGINAL LENGTH
1	1.0	--
4	1.618	61.80
5	2.618	38.20
6	4.24	23.61
7	6.854	14.59
8	11.090	9.02
9	17.944	5.57
10	29.034	3.44
11	46.979	2.13
12	76.013	1.32
13	122.992	.813
14	199.005	.502
15	321.997	.311
16	521.003	.192

* Reduction ratio $= \frac{\text{Final Range}}{\text{Initial Range}} = k$

This table can be used by the program user to estimate the number of iterations expected for a golden rule search. For example, if the initial range is 200,000 pounds and the final range is 5,000 pounds, the reduction ratio would equal 40. Table 12-1 indicates that 11 iterations will be required and that the final interval will only be 2.13% of the original interval of 200,000 pounds. The true optimum cargo weight/launch will be within 4260 pounds of the optimum value after 11 iterations. The actual calculation in the program uses equation 12.4-1

$$R = L_o G^{Q-3}$$

or

$$FR = (SPAN) \times G^{Q-3}$$

where

FR = final range in pounds (user input)

SPAN = UL - LL = difference between the upper and lower limits of the initial search range.

Solving for Q

$$Q = 3 + \frac{\ln(\frac{FR}{SPAN})}{\ln G}$$

where

$\frac{FR}{SPAN}$ is the reduction ratio used in Table 12-1. Number of iterations required is

$$Q = INT(Q + 1), INT = \text{integrize}$$

The number of iterations after the first 4 is

$$LIM1 = Q - 4 = INT\ (Q + 1) - 4$$

$$= INT\ (4 + \frac{\ln(\frac{FR}{SPAN})}{\ln G}) - 4$$

$$LIM1 = INT\ (\frac{\ln(FR) - \ln\ (SPAN)}{\ln G})$$

The program iterates LIM1 times after the first four evaluations.

The main assumption of the golden rule search procedure is that the function being examined is unimodal. For the case of total program cost versus cargo weight/launch this is a very tenable assumption. The minimum cost cargo weight/launch can occur either at the boundary points of the original search range or at some interior point. The golden rule search applies equally well to either possibility. Of course, the program user must be aware that if the selected optimum cargo weight is at one of the orignial boundary points, he has probably not included the true optimum cargo weight in his original search range. The convergency of the search to one of the boundary points indicates that the cost versus cargo weight was monotonically increasing or decreasing throughout the original search range.

Another limitation of the golden rule search occurs when the search has converged to a very small region about the true minimum. The slope in this region is very near zero and the search technique becomes very sensitive to small changes in total program cost. If a true functional relationship existed which was continuous throughout, no problem would exist. The cost versus cargo weight relationship is not continuous because of the influence of the inventory quantities. As the cargo weight/launch increases, the inventory quantities decrease in integer steps and the costs directly related to inventory quantities are decreased in a stepwise fashion. These cost steps do not disrupt the search procedure until the cost increments of the search iterations are similar in size to the cost steps described above. This usually only occurs when the difference in program costs between the highest and lowest cost of the four most recent evaluations is less than 1%. This obviously can only occur when the search interval is very small around the optimum cargo weight/launch. It is only at this level that the golden rule search begins to lose its efficiency.

If the golden rule search is not selected, the program does not use three of the user inputs:

UL - Upper limit in pounds of the initial search range for the golden rule

LL - Lower limit in pounds of the initial

FR - Final search range in pounds for a golden rule search

12.5 SIGNIFICANT VARIABLE NAMES

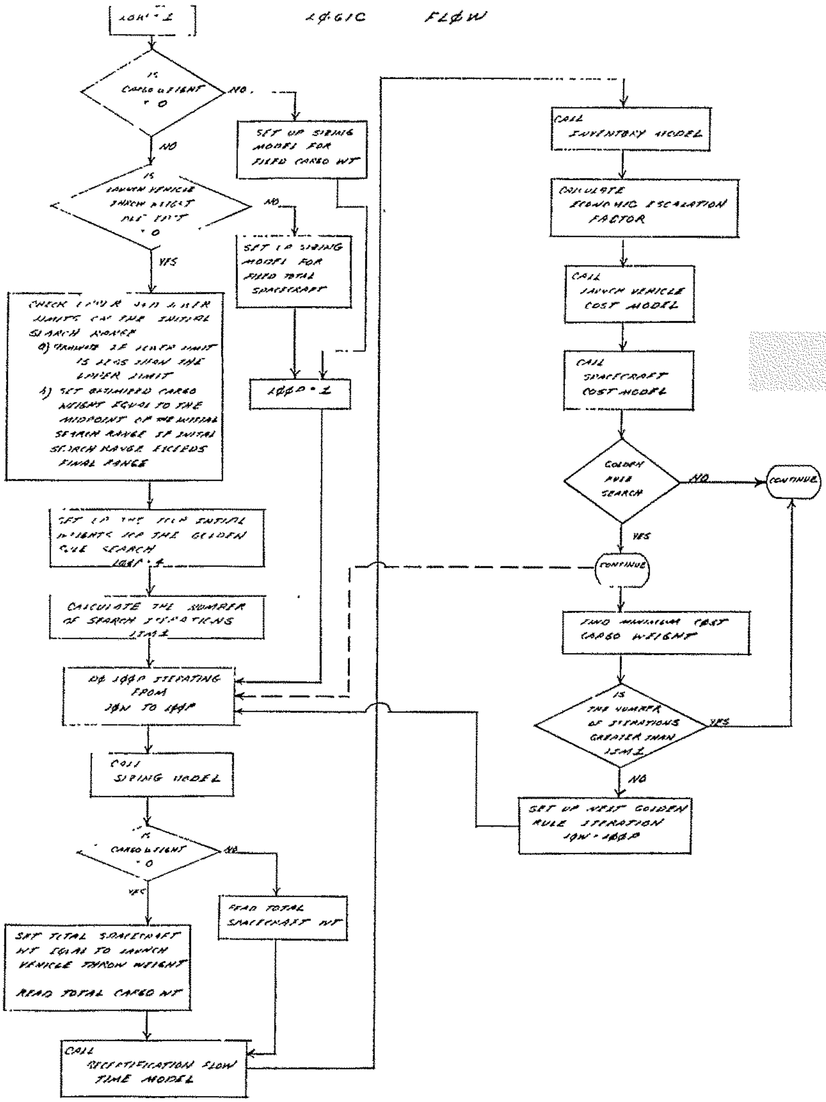
COUNT -	Counts number of iterations for a golden rule search
G -	Golden rule constant $G = 2/(1 + \sqrt{5}) \cong .618$
GOLD -	Golden rule indicator = 0 no golden rule = 1 golden rule
LIM1 -	Number of iterations for the golden rule search
LNG -	Natural logarthm of the golden rule constant
LOOP -	Upper limit on the cargo optimization DO loop
LOW -	Lower limit on the cargo optimization DO loop
LVTW -	Launch vehicle throw weight capability in pounds for the input orbit inclination and launch site
SCWT -	Total spacecraft weight in pounds, if Reuse = A, B, C use spacecraft effective weight if Reuse = D, E or F, use spacecraft gross weight
SMALL -	Total program cost in millions of dollars for the optimized cargo weight/launch
SPAN -	Range of the initial search interval in pounds
START -	Lower limit in pounds of the search range for each iteration
TPC(I) -	Total program cost in millions of dollars for the four current values of the golden rule search

WT(I) - Four current cargo weights/launch used in the golden rule search

12.6 CARGO WEIGHT OPTIMIZATION LOGIC FLOW - The cargo weight optimization logic flow is illustrated in Figure 12-3.

CARGO WEIGHT OPTIMIZATION LOGIC FLOW

FIGURE 12-3



13.0 OPERATIONAL VARIATIONS - The OCPDM program requires seven operations indicators from the program user. The seven indicators together comprise one set of operational variations. The user can input several sets of operational variations on the same run. The alternates available to the user for each operations indicator are indicated in Section 5.3.2 of this book.

When more than one set of operations is input, the program will run each set and select the optimized set on the basis of total program cost. The program will size the vehicle for each set according to the mode selected as described in Section 5.4.2 of this book. The following sections will examine each operations indicator and describe the specific program areas which are dependent on the indicator.

13.1 LAUNCH OPERATIONS INDICATOR - There are two options available to the program user:

LØ = 1 Gemini style

LØ = 2 Integrated checkout

The launch operations indicator appears in four CER's of the operations cost model.

a) Launch operations labor cost - boosted flight test (Volume II, Book 5 Section 6.2.15.1)
b) Launch area support labor cost - boosted flight test (Volume II, Book 5, Section 6.2.15.2)
c) Launch operations labor cost - operations phase (Volume II, Book 5, Section 6.4.1)
d) Launch area support labor costs - operations phase (Volume II, Book 5, Section 6.4.2)

If the integrated checkout option is selected (LØ = 2), the labor costs for the operations above will be 70% of the labor costs for the Gemini type launch operations.

13.2 AGE CONCEPT INDICATOR - There are two options available to the program user:

AGE = 1 semiautomatic

AGE = 2 onboard checkout

The AGE concept indicator appears in several operations CER's

a) Launch operations labor cost - boosted flight test (Volume II, Book 5, Section 5.2.15.1)

b) Launch area support labor cost - boosted flight test (Volume II, Book 5, Section 6.2.15.2)

c) Launch operations labor cost - operational phase (Volume II, Book 5, Section 6.4.1)

d) Launch area support labor cost - operational phase (Volume II, Book 5, Section 6.4.2)

e) Recertification labor cost - operational phase (Volume II, Book 5, Section 6.4.7)

The AGE concept indicator also affects the reusable vehicle turnaround time analysis. Two specific areas affected are:

a) Subsystem testing (Section 14, this book)

b) Prelaunch time (Section 14, this book)

For all the equations indicated above, the AGE concept is used to define a multiplying factor, AGEF, which is actually used in the equations. The AGEF factor for each AGE indicator is:

If AGE = 1, AGEF = 1.15
AGE = 2, AGEF = 0.95

If the onboard checkout system is selected (AGE = 2), an additional 149 pounds is added to the avionics weight for the entry vehicle. A fixed first unit cost of one million dollars and a development cost of 29 million dollars (1969 dollar base) is added to the total program cost to account for this additional system.

13.3 <u>REFURBISHMENT CONCEPT INDICATOR</u> - There are three options available to the program user:

REFP = 1 scheduled maintenance and testing with hot firing test (for REUSE = D, E or F)

REFP = 2 scheduled maintenance and testing with no hot firings (for REUSE = D, E or F)

REFP = 3 limited maintenance, routine postflight maintenance

The refurbishment concept indicator appears in the recertification labor cost equation defined in Volume II, Book 5, Section 6.4.7. Two specific areas of recertification are affected:

a) Subsystem testing

1) If REFP = 1 or 2, full subsystem checkout

2) If REFP = 3, limited subsystem checkout, ie. continuity testing only

b) Hot firing testing
   1) If REFP = 1, hot firing testing
   2) If REFP = 2 or 3, not hot firing testing

Obviously the hot firing testing only applies to integral propulsion vehicles. For a modular vehicle REFP = 1 and REFP = 2 will give identical results.

The refurbishment concept also affects the recertification flow time analysis for subsystem testing, upper stage propulsion testing and hot firing testing. If the limited maintenance option is selected, REFP = 3, the subsystem and upper stage propulsion test time in work days is only 1/3 of the complete maintenance and test time. If the hot firing test is selected, REFP = 1, for an integral propulsion vehicle, an additional 30 work days is added to the refurbishment cycle time.

A detailed explanation of the flow time analysis is given in Section 14.1 of this book.

This operational indicator is not used if an all expendable vehicle is being used.

13.4 <u>REFURBISHMENT SITE INDICATOR</u> - There are two options available to the program user:

REFS = 1 factory

REFS = 2 new site

If the factory is selected as the refurbishment site, REFS = 1, no additional facility costs are added to the launch site and recovery sites costs already calculated for the RDT&E phase. If a new site for refurbishment is selected, REFS = 2, an additional 250 million dollars (1969 dollar base) is added as additional facilities cost and included in the investment phase. Obviously for all expendable vehicles this indicator is not used.

13.5 <u>RECOVERY MODE INDICATOR</u> - There are two recovery modes available to the program user:

RECM = 1 water

RECM = 2 land

The water recovery mode is only appl[illegible] the all expendable ballistic vehicle. The program will use land for all other configurations regardless of the RECM value. Land landing can also be specified for all expendable ballistic.

Recovery Mode Indicator

RECM = 1 watch

= 2 land

If RECM = 1, ballistic vehicle uses a parachute for its descent to the water and no air drop test is required. If RECM = 2 is indicator, the vehicle uses a sailwing and an air drop test is required.

The water landing vehicle has less stringent design requirements for guidance and control and the G & C system weight is 345 pounds. The G & C weight for the land landing vehicle is 565 pounds. This increase in G & C will be reflected in both the RDT&E and investments costs. The larger G & C system will require more power and this is reflected in increased costs for electrical power. The land landing vehicle also carries additional weight for the skid gears needed for land landing.

The recovery mode indicator is used in the CER for recovery operations for both the boosted test flights (Volume II, Book 5, Section 6.2.15.6) and the operational phase (Volume II, Book 5, Section 6.4.6.). The recovery mode indicator is also used in the CER for recovery site facilities built during the RDT&E phase (Volume II, Book 5, Section 6.2.10.1).

13.6 RECOVERY SITES INDICATOR - There are four options available to the program user:

RECS - 1 - 2 new recovery sites

RECS = 2 - 2 existing recovery sites

RECS = 3 - 3 existing recovery sites

RECS = 4 - 4 existing recovery sites

The recovery sites indicator is used in the recovery site facilities CER defined in Volume II, Book 5, Section 6.2.10.1. There are no recovery site facilities for a water landing vehicle and consequently the recovery sites indicator is not used for a water recovery mode.

13.7 TRANSPORTATION MODE INDICATOR - There are three transportation modes available to the program user:

TRANS = 1 water

TRANS = 2 land

TRANS = 3 air

The trasnportation mode indicator is used in the transportation costs CER for both the RDT&E phase (Volume II, Book 5, Section 6.2.15.8) and the operational phase (Volume II, Book 5, Section 6.4.8).

The transportation mode indicator is also used to determine the transportation time for both reusable and expendable entry vehicles. The times in the OCPDM program are shown in Table 13-1.

TABLE 13-1

Transportation Time - days

Transportation Mode	Reusable Vehicle	Expendable Vehicles
Water	49	39
Land	16	11
Air	4	2

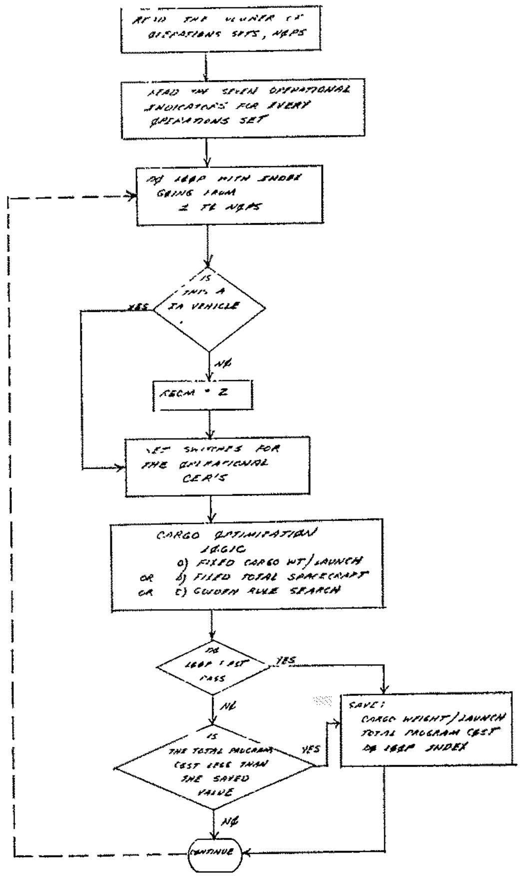
The times for reusable vehicles are larger since they include transportation to the refurbishment site and back to the launch site.

Also built into the OCPMD model are certain restrictions if the land or air transportation mode is selected. If the air mode is selected the maximum vehicle diameter is 25 feet and the maximum vehicle length of either the entry vehicle or the mission module is 70 feet; for land transportation the maximum diameter is 12 feet and the maximum length is 60 feet. If any one of these restrictions is exceeded, the program changes to the water transportation mode.

13.8 OPERATIONAL VARIATIONS OPTIMIZATION LOGIC FLOW - The logic flow for the operational variations optimization is illuestrated in Figure 13-1.

FIGURE 13-1

OPERATIONAL VARIATIONS
OPTIMIZATION
LOGIC FLOW



14.0 MISCELLANEOUS LOGIC

14.1 RECERTIFICATION FLOW TIME LOGIC - The objective of the recertification flow time subroutine, REFTIM, is to provide the inventory model with:

a) first unit total calendar days for recertification of the entry vehicle (for reusable entry vehicles)
b) transportation time
c) first unit prelaunch time

14.1.1 RECERTIFICATION TIME - The recertification flow time is the sum of six activities, conducted in series.

a) structure refurbishment
b) subsystem refurbishment
c) subsystem testing
d) refurbishment of the upper stage propulsion
e) upper stage propulsion testing
f) hot fire testing

The structure refurbishment time is a function of the ablative and radiative panel area. The structure refurbishment time in hours is:

STR = 120.2 + .05 (ABA) + .06 (RADA) where

STR = structure refurbishment time in hours

ABA = ablative panel area in square feet

RADA = radiative panel area in square feet

The subsystem refurbishment time for a ballistic vehicle is 448 hours and for a lifting body vehicle is 496 hours. In the OCPDM model two 8-hour shifts are assumed for a 5 day work week. Consequently, the total structure and subsystem refurbishment time is divided by 16 to obtain the number of work days.

The subsystem testing time in days is a function of both the AGE and refurbishment concepts. Table 14-1 indicates the total work days for the different combinations of refurbishment and AGE concepts.

TABLE 14-1
Subsystem Test Time

Refurbishment Concept \ AGE Concept	AGE = 1 Semiautomatic	AGE = 2 Onboard Checkout
REFP = 1 ØR = 2 Scheduled Maintenance	48.3 days	39.9 days
REFP = 3 Limited Maintenance	16.1 days	13.3 days

Obviously for vehicles which do not have integral upper stage propulsion, the refurbishment, test and hot firing time is zero. For vehicles with upper stage propulsion, the refurbishment time in work days, USP, is

USP = 6 X (No. of engines)

The test time and hot firing test time are both functions of the refurbishment concepts. The various possibilities are shown in Table 14-2.

TABLE 14-2
Test Time and Hot Firing Test Time for
Upper Stage Propulsion

Refurbishment Concept	Test Time (days)	Hot Firing Time (days)
REFP = 1 Scheduled maintenance Hot firing test	4 x (no. of engines)	30
REFP = 2 Scheduled maintenance No hot firing test	4 x (no. of engines)	0
REFP = 3 Limited maintenance No hot firing test	4/3 x (no. of engines)	0

After all the above activities have been calculated and added together, the sum represents the total work days for vehicle refurbishment. The total calendar days is obtained by multiplying the total work days by 7/5. In the inventory model a 90% learning factor is applied to the recertification time.

14.1.2 TRANSPORTATION TIME - The total transportation times used in the OCPDM model are shown in Table 14-3.

TABLE 14-3
Transportation Time

Transportation Mode \ Vehicle	Reusable	Expendable
TRANS = 1 Water	49 days	39 days
TRANS = 2 Land	16 days	11 days
TRANS = 3 Air	4 days	2 days

Built into the OCPDM model are certain size restrictions if the land or air transportation mode is selected. If the air mode is selected, the maximum vehicle diameter is 25 feet and the maximum vehicle length of either the entry vehicle or the mission module is 70 feet. For land transportation the maximum diameter is 12 feet and the maximum length is 60 feet. If any one of these restrictions are exceeded the program changes to the water transportation mode. If the transportation mode is changed internally, the output listing of the operational variations will indicate the change.

14.1.3 PRELAUNCH TIME - The first unit prelaunch time in calendar days is a function of

a) Configuration type
b) Reuse category
c) Total spacecraft length
d) AGE concept

The formula used in the OCPDM model is

$$TPL = A_F \times B_F \times \left(\frac{329.7}{348.5 - LENGTH}\right) \times (41 + 35 \times AGEF)$$

where:

TPL = first unit prelaunch time in calendar days

A_F = factor related to configuration

A_F = 1.00 if CONFIG = 1

A_F = 1.04 if CONFIG = 2

B_F = factor related to reuse category

B_F = 1.00 if REUSE = 1, 2 or 3

B_F = 1.10 if REUSE = 4, 5 or 6

LENGTH = total length of the spacecraft in feet

AGEF = factor related to the AGE concept

AGEF = 1.15 if AGE = 1

AGEF = 0.95 if AGE = 2

In the inventory model an 87.78% learning factor is applied to the prelaunch time.

14.1.4 MAIN VARIABLE NAMES - REFTIM SUBROUTINE

AF	Configuration factor in first unit prelaunch time calculation AF = 1.00 if CONFIG = 1 AF = 1.04 if CONFIG = 2
AGEF	AGE factor determined by the AGE indicator AGEF = 1.15 if AGE = 1 AGEF = 0.95 if AGE = 2
ATP	Refurbishment time in hours for the ablative thermal protection
BF	Reuse factor in first unit prelaunch time calculation BF = 1.00 if REUSE = 1, 2 or 3 BF = 1.10 if REUSE = 4, 5 or 6
CONFIG	Configuration indicator CONFIG = 1 - ballistic CONFIG = 2 - lifting body
CORE (4931)	Length in feet of the entry vehicle
CORE (5057)	Maximum diameter in feet of the entry vehicle
CORE (5168)	Length of the mission module in feet
DPR (136)	Number of engines for the upper stage propulsion
FTD	Refurbishment time in work days for the structure and subsystems
GEN (18,1)	Area of the radiative panels in square feet for the crew section
GEN (18,2)	Area of the radiative panels in square feet for the cargo/propulsion section
GEN (19,1)	Area of the ablative panels in square feet for the crew section
GEN (19,2)	Area of the ablative panels in square feet for the cargo/propulsion section
HFT	Work days for the hot fire testing

LENGTH	Length of the complete spacecraft in feet
MAXDIA	Maximum diameter in feet of the entry vehicle
MAXLEN	Maximum length of either the entry vehicle or the mission module in feet
REFP	Refurbishment concept indicator
REUSE	Reuse category indicator
RTP	Refurbishment time in hours for the radiative thermal protection
SS	Subsystem refurbishment time in hours
STR	Structure refurbishment time in hours
SWTRAN	Switch indicator for an internal change in the transportation mode SWTRAN = 0 no change SWTRAN = 1 change to transportation
TCD	Total calendar days for entry vehicle recertification
TPL	First unit prelaunch launch time in calendar days
TRANS	Transportation mode indicator
TT	Transportation time in days
TWD	Total work days for entry vehicle recertification
USP	Refurbishment time on work days for the upper stage propulsion
USPT	Test time in work days for the upper stage propulsion
VSST	Subsystem test time in work days

14.1.5 RECERTIFICATION TIME FLOW LOGIC - The logic for the recertification flow time model is illustrated in Figure 14-1.

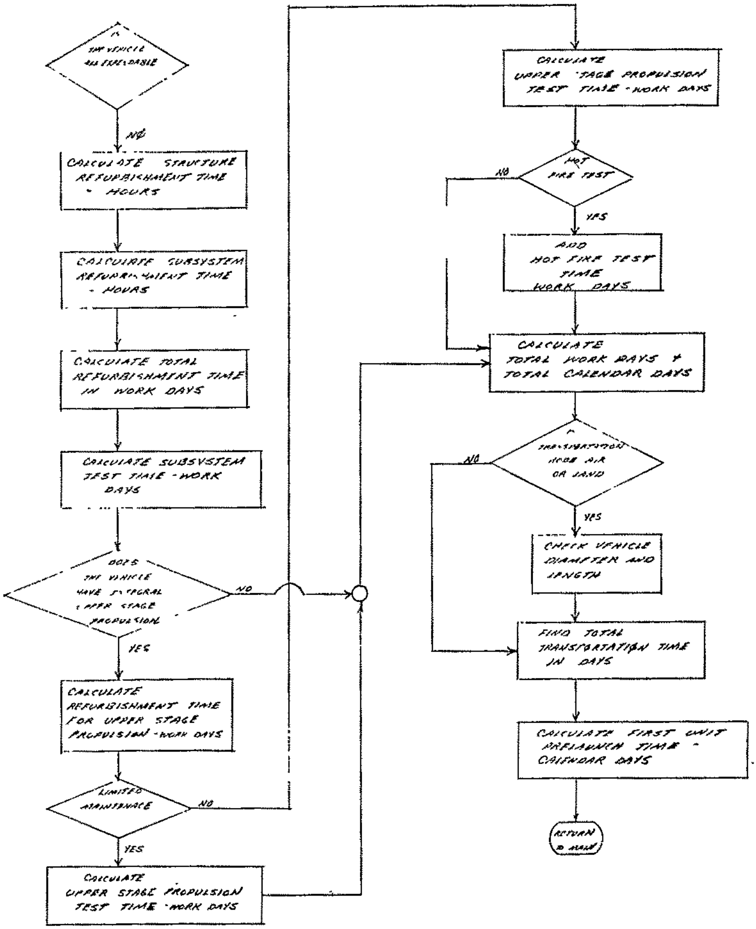
14.2 PROGRAM SCREENING LOGIC - The program user for the OCPDM model must input between 40 and 50 input values. It is possible to input a set of values which are not consistent. Although it is not possible to program the model to anticipate every inconsistency, many of the basic problem areas have been recognized and appropriate screening logic introduced to check for these inconsistencies and to printout appropriate error messages when applicable.

14.2.1 ERROR MESSAGES - There are several error messages which the program user may encounter. The program will immediately terminate after the message is printed out. Since the program terminates after the first error is

FIGURE 14-1

RECERTIFICATIØN FLØW TIME SUBRØUTINE

LØGIC FLØW



encountered, there may be other inconsistencies in the input data which will not appear.

a) "INCONSISTENT DATA"- This message will appear for any one of several possibilities dealing with the CWL, TCW, LR, PL and LVTWE input variables. The correct mode for inputting these parameters is given in Section 5.4.2 of this book.

1. LVTWE and CWL have both been defined (are non-zero). At least one must equal zero.
2. LR and PL are both zero. At least one must always be non zero.
3. TCW = 0 and either LR or PL equal zero. Only one of these three variables can equal zero in the same run.
4. Values of TCW, CWL, LR and PL have been defined and TCW ≠ (CWL) x (LR) x (PL)
5. Values for TCW, CWL, LR and PL have been defined and are consistent but PØMS (1) x PØMS (2) ≠ 1.0. The total mission success reliability must equal unity if the four parameters are input and consistent.

b) "LAUNCH VEHICLE NOT COMPATIBLE WITH SPACECRAFT CONFIGURATION"

This message is self-explanatory. The program user should check LVT and REUSE. The correct combinations are given in Section 8 of this book.

c) "CANNOT HAVE GOLDEN RULE WITHOUT SPECIFYING LVT"

Must input a specific launch vehicle when running a golden rule search.

d) "LAUNCH VEHICLE IS NOT COMPATIBLE WITH THROW WEIGHT REQUIREMENT"

The user has input a value of LVTWE which is greater than the thrown weight capability for the launch vehicle as defined in Volume 11, Book 5, Section 7.

e) "ORBIT INCLINATION CANNOT BE LESS THAN THE LAUNCH SITE LATITUDE"

The two launch sites used in the program are:

1. ETR - 28.5°N
2. WTR - 35.0°N

f) "ERRONEOUS VALUES INPUT FOR UPPER OR LOWER LIMITS TO GOLDEN RULE OPTIMIZATION"

The error message occurs if the upper limit is less than the lower limit.

14.2.2 DATA CHANGE LOGIC - Under certain conditions the program will change an input value when only one correct alternate exists. In the present program three input parameters may be altered internally.

a. If a land or air transportation mode is input and the program finds the vehicle too large for the mode the TRANS switch will be changed to a water mode.

b. If a water recovery mode is indicated for any vehicle other than an all expendable ballistic, the mode will be changed to land landing.

c. If the upper or lower limits specified for the initial golden rule search range are greater than or less than the thrown weight capability of the specified launch vehicle one or both of the limits will be changed to be compatible with the launch vehicle.

14.3 INPUT DATA ERRORS - Although Section 14.2 indicates that many of the data input inconsistencies are screened, there are many input parameters which are not screened. Two types of errors are possible:

a. Illogical value e.g. (PØMS (1) = 1.1)

b. Use of a user option not applicable to the configuration being examined, e.g., examining upper stage propulsion alternates for a modular configuration.

All the user inputs are listed in Table 14-4 with brief comments on the applicable values for each.

Table 14-4

User Data Inputs

Input Name	Definition	Real (R) Integer	Allowable Values	Comments
AGE	AGE concept indicator	I	AGE = 1 or 2	
CB	Dollar year base	I	CB > 0, integer	If CB = 1969, no economic escalation
CD	Cargo density	R	CD > 0	
CØNFIG	Configuration switch	I	CØNFIG = 1 or 2	If reuse = 6, CØNFIG = 2
CS	Crew size	I	CS $\geq$ 1, integer	
CWL	Cargo weight/launch	R	CWL $\geq$ 0	CWL $\neq$ 0, fixed cargo weight/launch
DL	Design Life			
ECS (I)	Environmental control system alternates	I	see Table 5-13 of this book	Not used if NALT (10) = 0
ECSR (I)	Lower and upper reliability limits for ECS	R	0 $\leq$ ECSR (I) $\leq$ 1 of this book	ECSR (1) $\leq$ ECSR (2), note (1)
ELECT (I)	Electrical power system alternates	I	See Table 5-9 of this book	Not used if NALT (5) = 0
FR	Final range of the golden rule search	R	0 $\leq$ FR $\leq$ (UL-11)	FR is only used if a golden rule search is used
GC (I)	Guidance and control alternates	I	See Table 5-10 of this book	Not used if NALT (7) = 0
GCR (I)	Lower and upper relaibility limits for guidance and control	R	0 $\leq$ GCR (I) $\leq$ 1	GCR (1) $\leq$ GCR (2), note (1)
HYD (I)	Hydraulic power alterantes	I	See Table 5-9 of this book	Not used if NALT (6) = 0

Table 14-4

User Data Inputs - con't

Input Name	Definition	Real (R) Integer	Allowable Values	Comments
INC	Orbit inclination	R	50°, 70°, 90°	
IR	Annual inflation rate	R	IR $\geq$ 0	
KENGR	Labor rate for engineering	R	KENGR > 0	
KLRS	Remote site labor rate	R	KLRS > 0	
KPRØD	Production labor rate	R	KPRØD > 0	
KTØØL	Tooling labor rate	R	KTOOL > 0	
LL	Initial lower limit for golden rule search	R	LL > 0	LL $\leq$ UL not used if golden rule is not used
LR	Annual launch rate	R	LR $\geq$ 0	
LS	Launch site indicator	I	LS = 1 or 2	
LVT	Launch vehicle type switch	I	LVT = 0, 1, 2, 3, 4 or 5	
LVTWE	Launch throw weight due east from ETR	R	LVTWE $\geq$ 0	
NALT (I)	Number of alternates for the Ith subsystem	I	NALT (I) = 0 or positve integer	See Tables 5-14 through 5-22 for the maximum NALT for each subsystem
NØPS	Number of operational sets	I	NØPS = positive integer	
ØACR (I)	Lower and upper reliability limits for the attitude control	R	0 $\leq$ Ø ACR (I) $\leq$ 1	ØACR (1) $\leq$ ØACR (2),Note (1)

Table 14-4

User Data Inputs - con't

Input Name	Definition	Real (R) Integer	Allowable Values	Comments
OM (I)	Orbital maneuver alternates	I	See Table 5-8 of this book	Not used if NALT (4) = 0
ØST	Orbit stay time	R	ØST > 0	
PHDELV	Phasing ΔV	R	PHDELV $\geq$ 0	
PL	Operational program length	R	PL $\geq$ 0	
PØMS (1)	Mission relaibility launch vehicle	R	0 $\leq$ POMS (1) $\leq$ 1	
PØMS (2)	Mission reliability - spacecraft	R	0 $\leq$ POMS (2) $\leq$ 1	
PØSR	Probability of successful recovery of the entry vehicle	R	0 $\leq$ POSR $\leq$ 1	
PØWR (I)	Lower and upper reliability limits for power	R	0 $\leq$ POWR (I) $\leq$ 1	P WR (1) $\leq$ POWR (2), Note (1)
PRINT (I)	Print switches for sizing and cost model	I	Print (I) = 0 or 1	
RACR (I)	Lower and upper reliability limits for reentry attitude control	R	0 $\leq$ RACR (I) $\leq$ 1	RACR (1) RACR (2), Note (1)
RECM	Recovery mode switch	I	RECM = 1 or 2	Only used for a IA vehicle
RECS	Recovery sites switch	I	RECS = 1,2,3 or 4	Not used if RECM = 1
REFP	Refurbishment concept switch	I	REFP = 1,2 or 3	Not used for an all expendable vehicle

Table 14-4

User Data Inputs - con't

Input Name	Definition	Real (R) Integer	Allowable Values	Comments
REFS	Refurbishment site switch	I	REFS = 1 or 2	Not used for an all expendable vehicle
REUSE	Reuse category of the entry vehicle	I	Reuse = 1,2,3, 4,5 or 6	If config = 1, reuse ≠ 6
SSREL	Overall subsystem reliability	R	0 ≤ SSREL ≤ 1	SSREL ≤ PØMS (2)
STEV (I)	Entry vehicle structure alternates	I	See Table 5-5 of this book	Not used if NALT (1) = 0
STGVEL	Staging velocity of the solid first stage launch vehicle	R	STGVEL > 0	
STMM (I)	Mission module structural alternates	I	See Table 5-6 of this book	Not used if NALT (2) = 0 Not applicable if REUSE = 3,4,5 or 6
TCW	Total cargo weight to orbit	R	TCW ≥ 0	
TELE (I)	Telecommunications alternates	I	See Table 5-11 of this book	Not used if NALT (8) = 0
TELER (I)	Lower and upper reliability limits for telecommunications	R	0 ≤ TELER (I) ≤ 1	TELER (1) ≤ TELER (2), Note (1)
TRANS	Transportation mode wswitch	I	TRANS = 1,2 or 3	
TPS (I)	Alternates for the thermal protection system	I	See Table 5-12 of this book	Not used if NALT (9) = 0
TPSR (I)	Lower and upper reliability limits for the thermal protection system	R	0 ≤ TPSR (I) ≤ 1	TPSR (1) ≤ TPSR (2), Note (1)

Table 14-4

User Data Inputs - con't

Input Name	Definition	Real (R) Integer	Allowable Values	Comments
UL	Initial upper limit for a golden rule search	R	UL $\geq$ 0	UL $\geq$ LL Not used if golden rule search is not used
USB (I)	Alternates for the upper stage boost propulsion	I	See Table 5-9 of this book	Not used if NALT (3) = 0 Not applicable if REUSE = 1,2 or 3
USER (I)	Lower and upper reliability limits for the upper stage boost propulsion	R	0 $\leq$ USBR (I) $\leq$ 1	USBR (1) $\leq$ USBR (2), Note (1)
VERNR (I)	Lower and upper reliabilities for the vernier maneuver	R	0 $\leq$ VERNR (I) $\leq$ 1	VERNR (1) $\leq$ VERNR (2), Note (1)

Note (1) If the lower and upper reliability limits for a subsystem are equal, the reliability for that subsystem is set equal to that limit.

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